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AIRPLANE ACTUA ION TRADE STUDY

T. R BOLDT
C. C. CH NOWETH
I. S. ME..DI
E. T. RAYMOND
L. WITONSKY
R. F. YURCZYK

BOEING MILITARY AIRPLANE COMPANY
SEATTLE, WASHINGTON 98124

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GREGORY J. CECERE
Program Manager
Control Techniques Group

PAUL D. LINDQUIST
Program Manager
Fluid Power Group

EVARD H. FLINN, Chief
Control Systems Development Branch
Flight Control Division

RICHARD D. FRANKLIN, Maj, USAF
Chief, Power Systems Branch
Aerospace Power Division

FOR THE COMMANDER

ERNEST F. MOORE, Col, USAF
Chief, Flight Control Division
Flight Dynamics Laboratory

FOR THE COMMANDER

JAMES D. REAMS
Chief, Aeros ace Power Division
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This report contains the results of The Airplane Actuation Trade Study program. The study, conducted in three phases, included establishment of actuation requirements; design of two airplanes (a baseline airplane and an all-electric airplane); and a trade study of the two airplanes including quantitative comparison data of weight, reliability and maintainability, and life cycle cost.

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The study results indicate that it is feasible to design a competitive power-by-wire actuation airplane. However, specific areas in hardware development need to be demonstrated through Research and Development programs to make the all-electric-airplane concept practical and low risk in the 1990+ time frame.

Cost savings were identified with the all-electric airplane for the AI mission. These came primarily from reduced secondary power system weight and complexity at some expense in ground checkout capability without running the engines. This study selected engine-shaft-mounted electric generators as opposed to airframe mounted accessory drives for the baseline airplane.

Different mission/air vehicles will have to be studied individually to project cost and other benefits available from an all-electric airplane concept. The AFS study showed minor differences between hydraulic and all-electric applications, with a slight advantage toward an all-electric approach in terms of life cycle cost.

PREFACE

This report describes the work performed by The Boeing Military Airplane Company, Advanced Airplane Branch, Seattle, Washington on an airplane actuation trade study. This work was sponsored by the Air Force Wright Aeronautical Laboratories, Flight Dynamics Laboratory and Aero Propulsion Laboratory, Wright-Patterson Air Force Base, Ohio. Work was authorized under contract F33615-79-C-3630, Project No. 2403, work unit 24030. Gregory J. Ceccre of the Flight Dynamics Laboratory, Flight Controls Branch, AFWAL/FIGLA and Paul Lindquist of the Aero-Propulsion Laboratory, AFWAL/POOS-1 were the Air Force Project Managers. (At the initiation of the program the FDL project engineer was Mr. Daniel K. Bird who has since retired.)

Roger F. Yurczyk served as Program Manager and Ishaque S. Mehdi served as Principal Investigator.

The technical effort was performed by the following team of engineers:

Function	Responsible Engineers
Airplane Configuration and Requirements	C. C. Chenoweth
Flight Control Actuation Systems	C. C. Chenoweth J. S. Shen
Hydraulic Power System and Non-Flight Control Actuation Systems	E. T. Raymond H. F. Hillman
Electrical Power System	E. T. Reiquam I. S. Mehdi
Wheel Brakes and Steering System	R. H. Berg W. Mo[illegible]
Environmental Control System	R. M. J[illegible]nson
[illegible] Cooling System	J. [illegible] Runnels

Actuation Systems Structural Integration	E. D. Painter
Reliability and Maintainability	C. L. Swindell
Life Cycle Cost	L. Witonsky

In addition to this effort performed by the Boeing team, a study of electromechanical actuation systems for the All-Electric Airplane was conducted by the AiResearch Manufacturing Company of California under subcontract No. G-A87756-9176. The technical effort was conducted by Mr. Stephen Rowe and is reported in Reference 1.

The following companies provided information for this study at no cost to the program and their support is gratefully acknowledged.

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Inland Motor Division, Kollmorgen Corp., Radford, Virginia

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SUMMARY

The objective of this program was to establish the advantages/disadvantages and life cycle cost impact for two types of 1990+ time frame airplanes, one which has hydraulically powered actuation systems (Baseline Airplane) and the other which has electrically powered actuation systems (All-Electric or Power-By-Wire Airplane). A secondary objective of this program was to identify the 1990+ technology needs and development requirements of hydraulic, power-by-wire actuation systems and secondary power systems for future aircraft. The comparison was made of both the actuation and the secondary power systems. Parameters that were quantified for comparison were weight, reliability/maintainability and life cycle costs. In addition, qualitative evaluations were made on the basis of structural integration, growth potential, survivability/vulnerability, EMC/lightning, environmental constraints and technology risks.

The study was conducted in three phases. In Phase I, Development of Design Data Base, an air-to-surface (ATS) airplane configuration was established, the actuation functions were defined, and the requirements for these actuation systems were established.

The study was conducted using the Boeing Model 987-350 ATS as the point of reference airplane for which engineering development would begin in 1990, production in 1995, and initial operational capability (IOC) in 1997. The model 987-350 has an all-moving canard, an arrow wing, twin pod-mounted engines with variable geometry inlets and two-dimensional vectoring and thrust reversing nozzles, a thrust-to-weight ratio of 0.87 and a maximum gross weight of 49,000 lbs. The airplane carries an internally mounted 25 mm gun and 5000 lbs of air-to-ground weapons. The airplane is designed for a high level (Mach 2.2) and a low level (Mach 0.9 to 1.2) interdiction mission. The design life is 10,000 flight hours and 6,000 landings.

The actuation functions defined were flight controls (canard, elevons, rudder, spoilers and leading edge flaps), engine controls (inlet centerbody and bypass doors), landing gear (retraction, steering and brakes), aerial refueling (door

and nozzle latch), and canopy. Thrust vectoring/reversing actuation was determined to be pneumatic in the high temperature environment of that application, and therefore was not part of the hydraulic/electric actuation trade study. In addition, drive power for the 25-mm gun and environmental control system (boost and pack compressors, and cooling fan) was included. The actuation requirements were defined in sufficient detail so that systems for both the Baseline and All-Electric Airplanes could be designed.

An electrical load analysis was also prepared. The load analysis included the normal housekeeping and avionics electrical loads along with power requirements for actuation systems.

In Phase II, Design of Two Airplanes, the actuation and secondary power systems were designed for the Baseline and All-Electric Airplanes. Several configurations for each actuation function were developed and the optimum system was selected based on weight, envelope for structural integration, efficiency, power demand, system complexity and technology projections into the 1990's. The design and selection of the actuation systems for the All-Electric Airplane were primarily conducted with data supplied by the AiResearch Manufacturing Company of California under a subcontract. The power demands were determined for the hydraulic and electrical systems for the Baseline Airplane and for the electrical system for the All-Electric Airplane. Several secondary power system configurations were developed for both airplanes and an optimum system selected for each.

In Phase III, Trade Study, data for systems weights, reliability/maintainability, and life cycle costs were developed.

The reliability was computed by defining the minimum equipment levels for loss of mission and loss of aircraft, developing the fault trees and computing the probabilities.

The maintainability and life cycle costs were determined using the RCA PRICE and PRICE L computer programs. Each system (actuation and secondary power) for both airplanes was broken down to the line replaceable unit (LRU) and various input parameters were developed describing the quantity, weight, ratio of structure and electronics, complexities, and development and production dates. The output fro the PRICE program provided mean-time-between-failure (MTBF), development costs, and production costs. The PRICE L program also provided the Operating and Support Costs.

Based on the above data, overall weights, reliability/maintainability and life cycle costs were computed and compared. Along with this a qualitative assessment of the structural integration, growth potential, survivability/vulnerability, EMC/lightning, environmental constraints, and technology risk of the actuation and secondary power systems of both airplanes was conducted.

The results of this program indicate that the All-Electric Airplane offers a potential for reducing the life cycle costs of the actuation and secondary power systems by approximately 12% compared to the Baseline Airplane configuration. On an airplane of this type and size the weight penalty associated with EM actuation with respect to hydraulic actuation is offset by the weight savings in the secondary power system. The secondary power system for the All Electric Airplane uses engine-shaft mounted main AC generators as opposed to the AMAD concept for the Baseline Airplane. This results in reduced ground checkout capability for monitoring the main generator without running the engines.

The probabilities of mission success and airplane safety are comparable for both airplanes. The MTBF of the EM actuation system was lower than the hydraulic actuation; the MTBF of All-Electric secondary power system was higher than the conventional mixed hydraulic/electric secondary power system, but not enough higher to completely offset the lower MTBF of EM actuation.

Assessment of the other factors indicated that EM actuation and electrical secondary power system could ease structural integration problems and provide additional growth potential. From a survivability/vulnerability standpoint the hydraulic power system was more vulnerable than the electrical system from weapons effects, whereas the EM actuation system was more vulnerable to jamming due to the necessity of gearboxes in every application. EMC/lightning effects could impact the fly-by-wire (FBW) and electrical systems in either airplane, but the EM actuation would also be impacted in the All-Electric

Airplane. There were no high technology risks associated with the Baseline Airplane.

The study also indicated that a hybrid system arrangement may have some benefit. The results show that the primary payoff for the All-Electric Airplane resulted from elimination of the engine driven hydr ulic system, i.e., adapting a single source p er system. These benefits could also be realized through the application of integrated actuator packages (IAP), electr c motor driven hydraulic actuator systems. These showed some potential benefit for certain flight control functions. For example, the study results indicated that use of an IAP for rudder a tuation offered no weight penalty over the EM actuator and has a lower development risk.

The results and conclusions drawn from this study are based on an assumption that certain technology advancements will be made by the 1990+ time frame. Technology developments that are required to meet these needs or that offer alternatives in the design of the actuation and secondary power systems were identified. For the Baseline Airplane these include:

- o High pressure hydraulic system
- o Bi-directional power transfer units
- o Hydraulic fuses and circuit breakers
- o Load adaptive/stored energy actuators
- o Advanced fly-by-wire actuators
- o Staged sequential servo ram actuation

For the All-Electric Airplane the technology needs include the development of:

- o Lightweight, high efficiency gearboxes
- o Speed optimized electric motors
- o Load-adaptive/stored energy actuation techniques
- o Variable authority EM actuators
- o Controller/inverters
- o High voltage DC electric systems
- o Integrated actuator packages

Several of these developments identified for both the Baseline and the All-Electric Airplanes are applicable to a hybrid system.

I INTRODUCTION

1.1 Background

Current aircraft are characterized by having two main forms of on-board secondary power generation, distribution, and utilization, i.e., electrical power and hydraulic power. In general, hydraulic power is generated, distributed, and utilized for the majority of the actuation jobs including flight control surfaces, landing gear extension and retraction, brakes, and nose wheel steering. Electrical power is used for functions like stability augmentation, fuel and engine control, heating and cooling, lighting, avionics, weapons control, instrumentation, and utility air vehicle functions. Powered actuation is essential in today's high-performance aircraft. Landing gear, gun drive, and canopy operation also require high power. Superior airplane controllability and handling qualities characteristics require not only high power, but also accurate and responsive controls. Hydraulic actuation has become the mainstay for most of these control tasks because of high torque-to-inertia capability, high power and weight efficiency, and tremendous development and experience. Technology advancements in the electromechanical field are showing promise for alternative means of actuation. Consideration needs to be given and evaluations made with these new technology trends in mind.

Major factors stimulating the application of power-by-wire actuation are in the advancements in high-voltage power supplies, rare earth permanent magnet motors, electronic commutation, and improved solid-state power switching devices. These factors lead to the objectives of this study which are:

(1) Establish advantages/disadvantages and life cycle cost impact of hydraulically powered actuation and electrically powered actuation for aircraft in the 1990+ time frame.
(2) Identify technology needs, risks, and development requirements for future aircraft actuation systems.

1.2 Objective

The objective of this study was to conduct a trade-off comparison between a

"Baseline Airplane" (one that contains an engine-driven hydraulic system for actuation) and an "All-Electric Airplane" (one that contains only an engine-driven electrical system for power-by-wire actuation). The study was conducted on an ATS airplane. The airplane is designed for a high survivability interdiction mission. For the trade, each "airplane" is designed to utilize every beneficial technology advancement considered available in the 1990+ time frame. Six areas of actuation were considered in the study. These were the flight controls, engine inlet controls, thrust reverser/vector controls, landing gear, aerial refueling, and canopy actuation. In addition the gun controls and ECS were considered as users of secondary power.

1.3 Approach

The program was divided into three phases as follows:

Phase I - Development of ATS Design Data Base
Phase II - Design of Two Airplanes
Phase III - Airplane Actuation Trade Study

Baseline Airplane

The hydraulic/electric powered airplane was termed the Baseline Airplane. The hydraulic actuation systems considered various types of power drive units, output mechanisms, and control valving. Secondary power extraction is accomplished by power take-off shafts from each engine which drive airframe mounted accessory drives (AMAD). The two AMADs are connected together and to a LOX/JP-4 Integrated Power Unit (IPU) through an angle gearbox. During normal flight, the AMADs are driven by their respective engines and the angle gearbox is declutched. During an emergency, shaft power can be extracted from the opposite engine or the IPU through the angle gearbox. Each AMAD drives two hydraulic pumps and an electrical generator. The right-hand AMAD also drives the ECS boost compressor. This AMAD configuration provides the capability to operate the engine driven secondary power system without operating the engines, for ground checkout.

All-Electric Airplane

Two types of actuation systems were considered for the All-Electric Airplane actuation functions: electromechanical actuation (EMA) systems and integrated actuator package (IAP) systems. EMA's were selected for all functions since they proved lighter and less complex in all cases when compared with the equivalent IAP. Secondary power extraction is accomplished by a 150-kw starter-generator mounted on the spinner at the front of each engine. A third 150-kw generator is mounted on the LOX/JP-4 Integrated Power Unit (IPU). The three generators produce wild frequency power which is converted to 270V dc by phase delay rectifier (PDR) bridge converters. Secondary converters provide power at other voltages required. Interconnection provisions are included in the three generation systems for engine starting and transfer of loads in case of failure of the main generation systems. This system provides for ground checkout of all electrical functions, except the engine-driven generators/ regulators themselves, without operating the engines.

Trade Study

Ten parameters were considered in the trade study of the two airplanes:

- Weight
- Reliability
- Maintainability
- Life Cycle Costs
- Structural Integration
- Growth
- Survivability
- EMC/Lightning Protection
- Environmental Constraints
- Technology Risk

Quantitative comparison data were developed for the first four parameters. Qualitative comparisons were made in the six other areas.

II AIRPLANE REQUIREMENTS

The basic airplane configuration and requirements which formed the design data base for the trade study airplane were developed during Phase 1. Design criteria and requirements for the actuation functions and other functions requiring on-board generated secondary power were defined.

2.1 Airplane Configuration

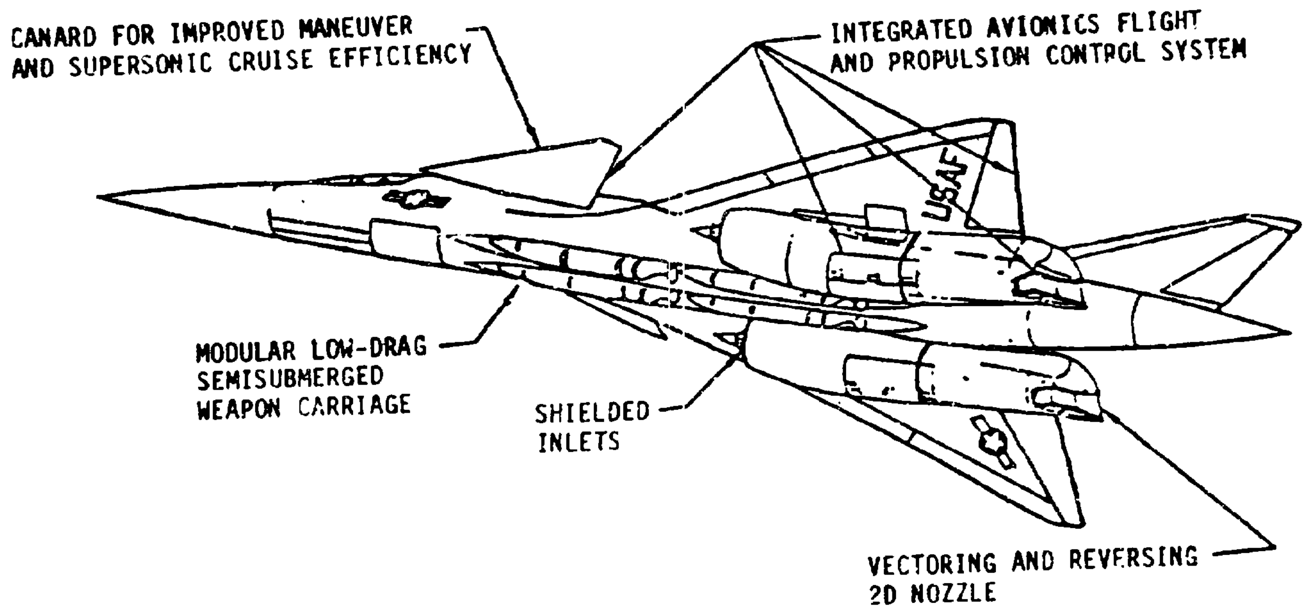
The ATS mission concept was specified as the point-of-reference airplane. The Boeing Model 987-350 ATS (Air-to-Surface) Airplane (Figure 1) was chosen for this purpose. It is a vectored-thrust, canard/arrow wing with a thrust-to-weight ratio of 0.87 and a gross weight of 49,000 lbs. The airplane configuration includes twin pod-mounted engines, wing-shielded half-round variable-geometry inlets, 2-D vectoring and thrust reversing nozzles, and an all-moving canard. Armament consists of an internally-mounted 25-mm gun, two advanced short-range missiles, and 5000 lbs of air-to-ground weapons mounted semisubmerged in two fuselage cutouts. Airplane performance is shown in Figure 2. STOL take-off and landing performance is shown in Figure 3. The airplane is designed for a high-survivability interdiction mission (Figure 4). The flight envelope is shown in Figure 5. Design life of the airplane is 10,000 flight hours and 6,000 landings.

2.2 Actuation System Requirements

The ATS Model 987-350 actuation system requirements were divided into five areas as follows:

- Flight Controls
- Engine Inlet
- Landing Gear
- Aerial Refueling
- Canopy Actuation
- Thrust Reverser/Vector Controls

It was determined that the thermal environment for the thrust reverser and



• GROSS WEIGHT	49,000 lb
• SPAN	30 ft
• LENGTH	80 ft
• T/W	0.87
• W/S	83
• FUEL	19,400 lb
• DESIGN PAYLOAD	5,000 lb

Figure 1 Model 987-350 Airplane

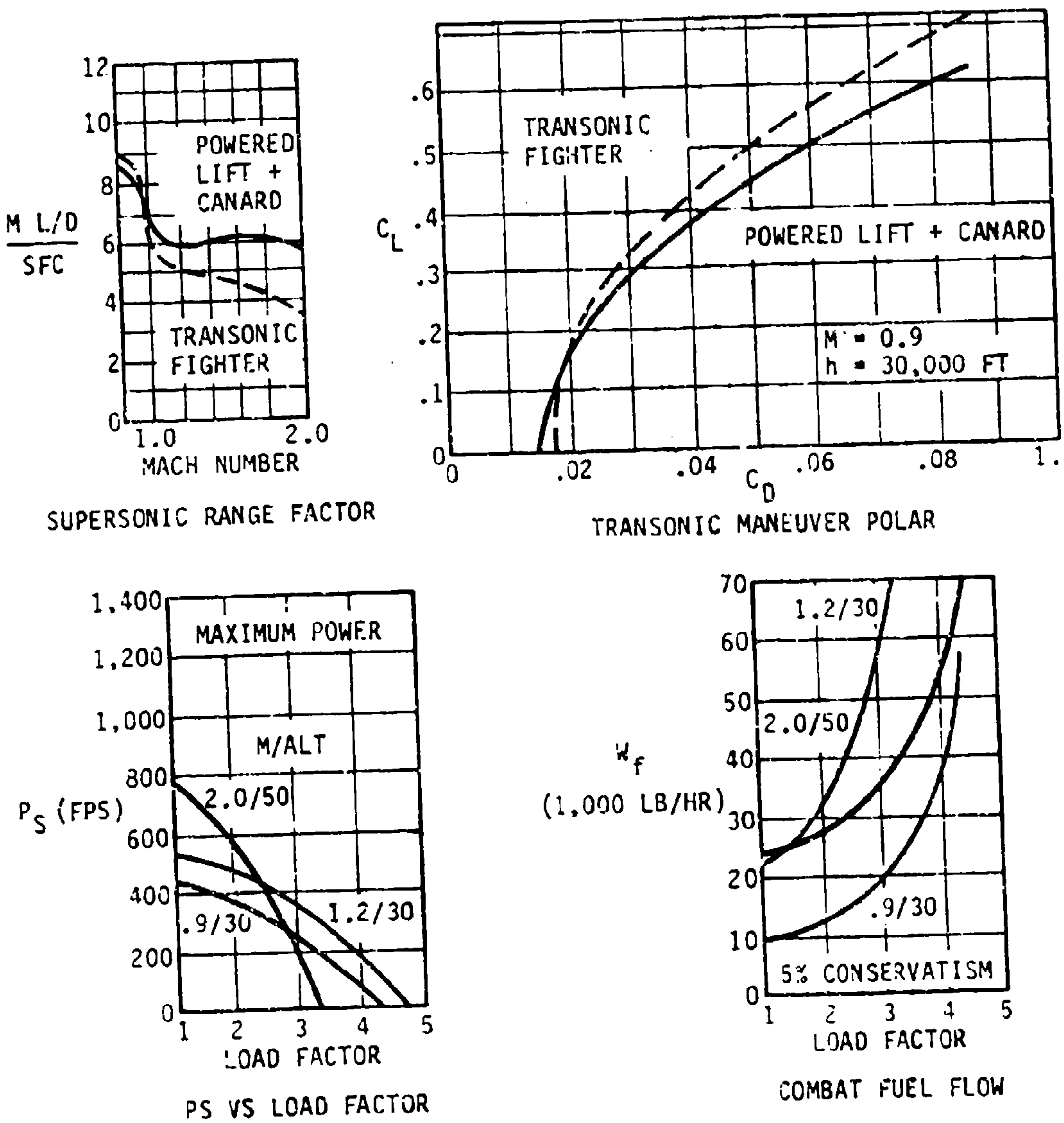


Figure 2 Model 987-350 Performance

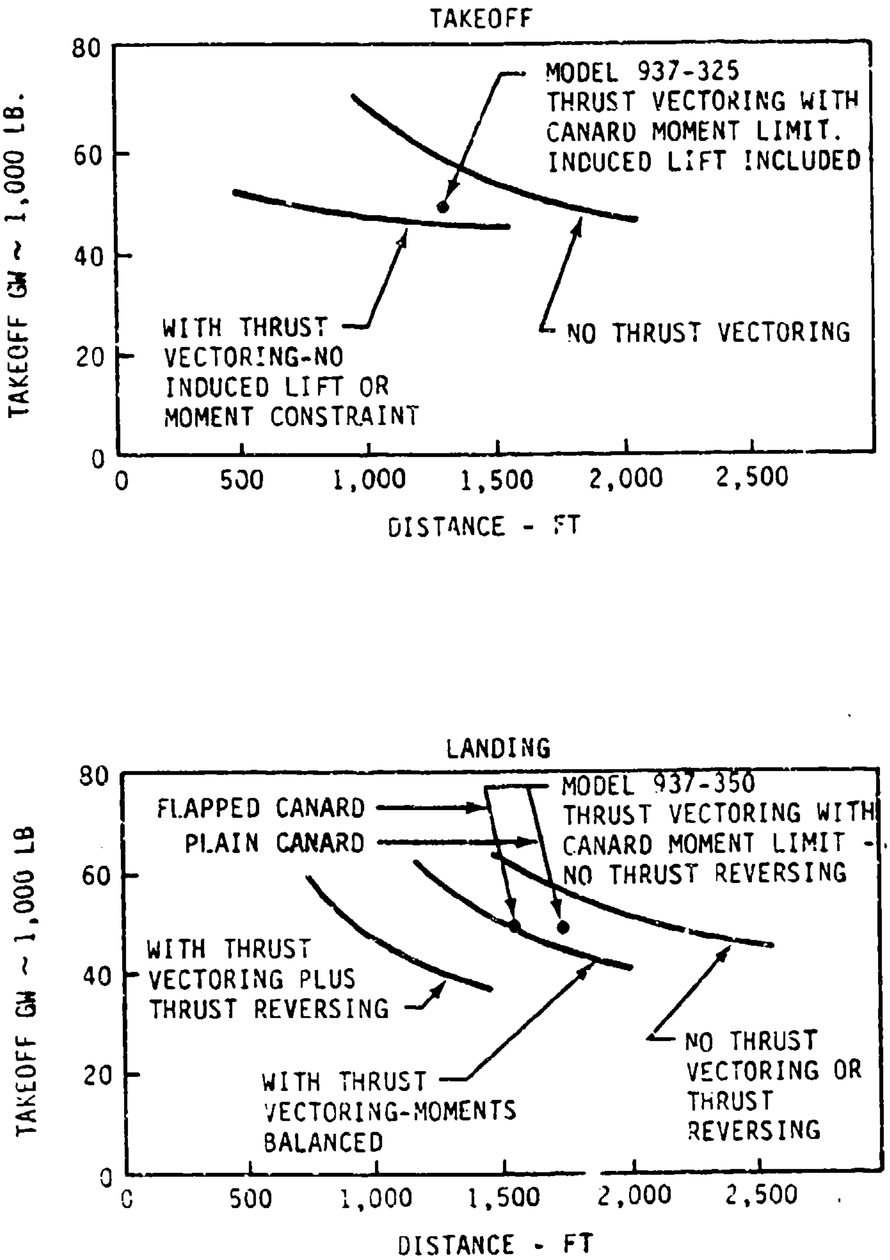
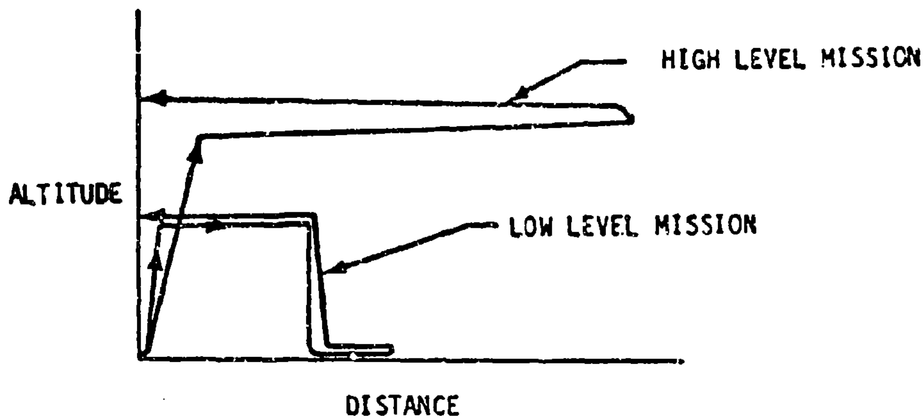


Figure 3 Model 987-350 STOL Performance



HIGH LEVEL MISSION

	ALT FT	DIS NMI	M
TAKEOFF, CLIMB	0	0	0-.9
CLIMB & ACCEL	0-63.700	33	0.9-2.2
CRUISE	63.700	394	2.2
180^{0} TURN	63.700	-	2.2
RETURN	69000-0	100%*	2.2-0

LOW LEVEL MISSION

TAKEOFF, CLIMB	0	0	0-.9
CLIMB	0-35000	11	0.9
CRUISE	35000-37000	140	.9
SUPERSONIC DASH IN	0	61	1.2
180^{0} TURN	0	-	1.2
RETURN	0-43000-0	50%*	1.2 - 0.9-0

* PERCENT OF SHORT RANGE COMBAT RADIUS

Figure 4 Model 987-350 Mission Profiles

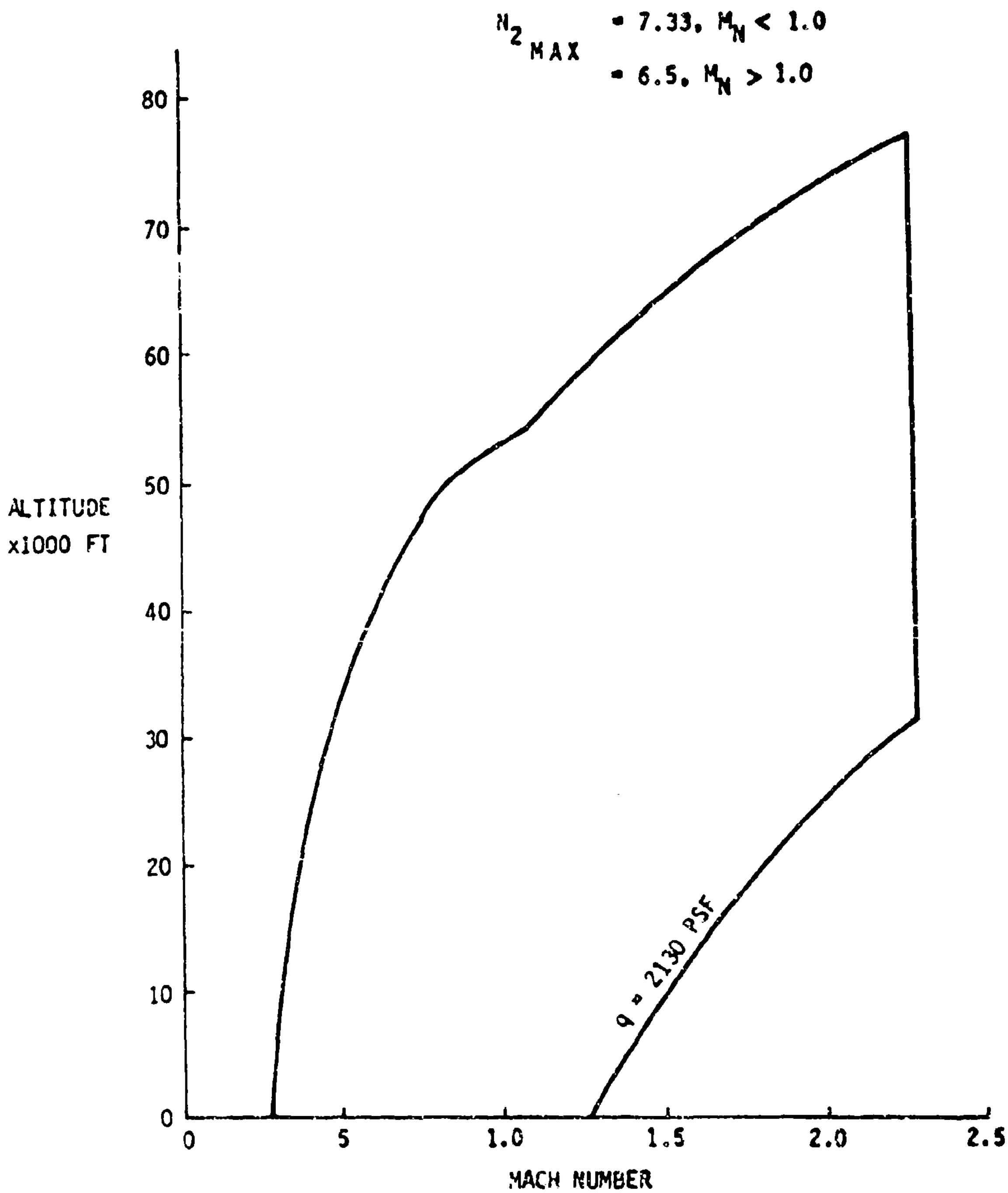


Figure 5 Model 987-350 Flight Envelope

vectoring actuation systems would be too harsh (Figure 6) for use of electromechanical or hydraulic actuators without auxiliary cooling provisions. Thus it was concluded that neither the electromechanical nor the hydraulic actuators could effectively compete with pneumatic actuators, traditionally used in these applications. These high temperatures can damage insulation on electric motor windings, would be close to the Curie temperature of the permanent magenets causing demagnetization, and cause motor bearing lubricant problems. In the case of hydraulic actuators, conventional hydraulic fluids could not be used and seal problems would also be encountered. To utilize electromechanical or hydraulic actuators would require either one or both of cooling provisions and remote location of actuators with complex mechanical linkages to transmit the actuation forces. This would add to the system complexity and impact the reliability and cost of the system. Therefore, pneumatic actuation systems for the thrust reversing and vectoring functions were selected. This allowed the deletion of these actuation functions from further consideration in this study.

In each of the other areas the number of actuators required for each function and the configuration and redundancy of the actuation systems were defined. The requirements are summarized in Tables 1 to 4.

2.3 Gun and ECS Power Requirements

Two additional areas where shaft power is utilized are the 25-mm gun system and the environmental control system.

The gun system requires 14 hp for the gun drive and 11 hp for the ammunition feed system. This power can be delivered by an electrical motor or hydraulic motor. The motors require start-up and reversing capability for shell clearing purposes. Figure 7 shows the power and speed vs firing rate.

The ECS, shown schematically in Figure 8, requires three motors; one each for the boost compressor, the ECS compressor and the ECS fan. The boost compressor motor has to provide 50.5 hp at speeds varying from 15,000 to 40,000 rpm to be compatible with the following boost compressor requirements:

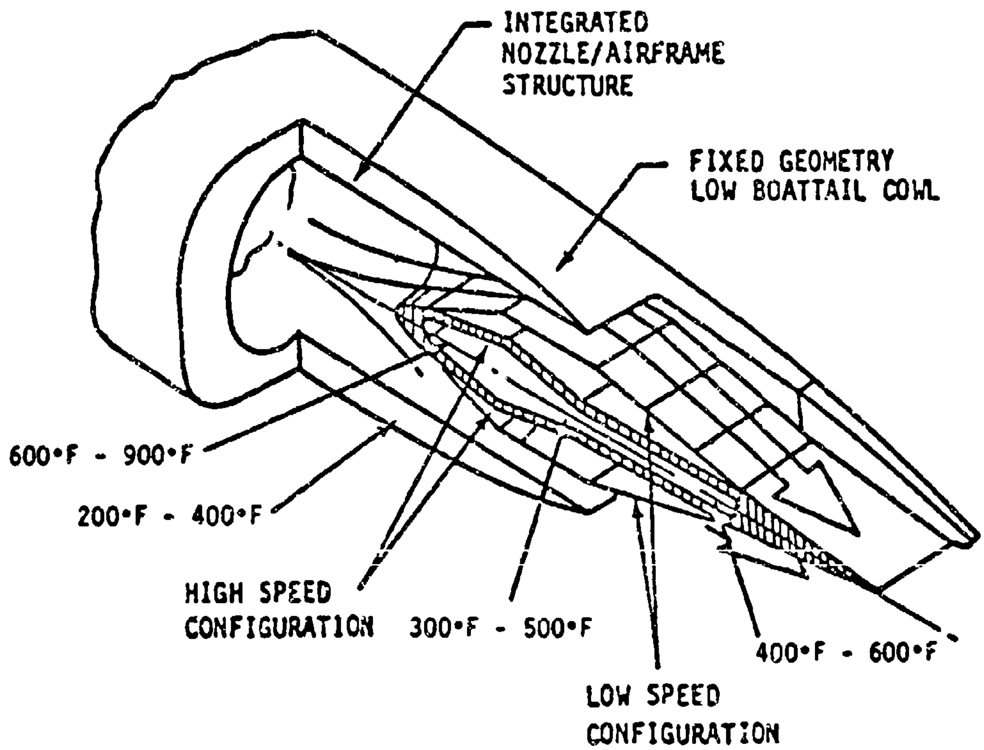


Figure 6 Engine Exhaust Area Temperatures

TABLE 1 SUMMARY OF ACTUATION REQUIREMENTS - CONTROL SURFACE

ACTUATOR	MAX TORQUE (STALL) FT-LB	MAX. RATE (NO LOAD) DEG/SEC	MAX POWER		(NOTE 1) MAX BANDPASS (FREQUENCY RESPONSE) SEC^{-1}	(NOTE 2) DUTY CYCLE CONTINUOUS	RESOLUTION DEGREE	HYSTERESIS DEGREE	THRESHOLD (FREEPLAY) DEGREE	MAX TRAVEL DEGREE	STATIC STIFFNESS IN-LB/RAD $\times 10^6$	DYNAMIC STIFFNESS IN-LB/RAD $\times 10^6$	LOAD INERTIA LB-IN^2	(NOTE 2) RESPONSE TO FAILURE	LIFE HOURS	MAX COMPARTMENT TEMPERATURE
			HP	CONDITION												
CANARD	+43,250 TOTAL	70	40 TOTAL	35000 FT.LB. X 35 DEG/SEC	20	10% RATE 10% LOAD	.03	0.1	.035	+ 20 - 10	29.7 TOTAL	29.7 TOTAL	53',400 TOTAL	FAIL-OP[2]	10,000	250° F
ELEVON	+19,200 -42,000 PER ELEVON	70	37 PER ELEVON	33000 FT.LB. X 34 DEG/SEC	20	15% RATE 20% LOAD	.04	0.1	.07	± 20	78.6 PER ELE.	14.04 PER ELE.	50,340 PER ELE.	FAIL-OP	10,000	250° F
RUDDER	±17,618 TOTAL	75	13 TOTAL	10000 FT.LB. X 43 DEG/SEC	20	10% RATE 5% LOAD	.06	0.25	.08	± 30	12.1 TOTAL	5.6 TOTAL	27,040 TOTAL	FAIL-OP	10,000	250° F
SPOILER	+3,534 -7,550 PER SIDE	100	11 PER SIDE	3435 FT.LB. X 100 DEG/SEC	20	5% RATE 20% LOAD SUBSONIC	.06	0.3	.1	60	1.18 PER SIDE	0.71? PER SIDE	6870 PER SIDE	FAIL-SAFE	10,000	250° F
L.E. FLAP	-101,400 TOTAL	15	41 TOTAL	86190 FT.LB. X 15 DEG/SEC	20	20% RATE 50% LOAD SUBSONIC	.06	0.3	.1	30	69.4 TOTAL	6.08 TOTAL	29,330 TOTAL	FAIL-OP	10,000	250° F
INLET CENTERBODY	20,800 LB PER CENTERBODY	4 IN/SEC	13 PER CENTERBODY	20800 LB. X 4 IN/SEC	10	20% RATE 50% LOAD SUPERSONIC ONLY	.004 INCHES	.04 INCHES	.01 INCHES	4 INCHES	400,000 LB./IN.	-	75 LB. PER CENTERBODY	FAIL-SAFE	10,000	250° F
INLET BYPASS DOOR	25 PER DOOR	90	0.07 PER DOOR	25 FT.LB. X 90 DEG/SEC	10	20% RATE 50% LOAD SUPERSONIC ONLY	.09	.9	.023	90	-	-	-	FAIL-SAFE	10,000	250° F

NOTE 1: PHASE SHIFT ≤45° FOR ± 1° AMPLITUDE
NOTE 2: SEE SECTION 3.3.1 FOR DETAILS

TABLE 2 SUMMARY OF ACTUATION REQUIREMENTS - LANDING GEAR

ACTUATOR	MAX TORQUE (STALL)	MAX RATE (NO LOAD)	MAX HP	MAX BANDPASS (FREQUENCY RESPONSE)	DUTY CYCLE	MAX TRAVEL	RESPONSE TO FAILURE
NOSE GEAR EXT-RET ACTUATOR	15,000 LB 10,000 FT-LB	2.5 IN/SEC 2.5 RPM	5.69	NOT APPLICABLE	ONCE PER FLT.	15 IN.	OPERABLE AFTER ONE POWER FAILURE
MAIN GEAR EXT-RET ACTUATOR	15,000 LB 10,000 FT-LB	2 IN/SEC 2.5 RPM	4.76	NOT APPLICABLE	ONCE PER FLT.	12 IN.	OPERABLE AFTER ONE POWER FAILURE
NOSE GEAR STEERING ACTUATOR	16,667 IN-LB	2 RPM	0.55	15 Hz.		$\pm 102^{\circ}$	FREE TO CASTER
	TORQUE	ENERGY (FT-LB)	VELOCITY	CONDITION			
MAIN GEAR BRAKES	15,000 FT-LB	——	——	PARKING	22 SECONDS PER LANDING		OPERABLE AFTER ONE POWER FAILURE
	10,700 FT-LB	29.2×10^6	164 KTS	RTO			
	8,680 FT-LB	19.4×10^6	148 KTS	MAX LDG			
	6,400 FT-LB	10.8×10^6	128 KTS	NORM LDG			

TABLE 3 SUMMARY OF ACTUATION REQUIREMENTS - AERIAL REFUELING & CANOPY

ACTUATOR	MAX TORQUE (STALL)	MAX RATE (NO LOAD)	MAX HP	MAX BANDPASS (FREQUENCY RESPONSE)	DUTY CYCLE	MAX TRAVEL
AERIAL REFUELING ACTUATORS						
DOOR ACTUATOR-LINEAR	3280 LB LOAD	0.4 IN. PER SEC	0.19	NOT APPLICABLE	ONCE PER FLT.	2.35 IN.
NOZZLE LATCH ACTUATOR-LINEAR	1750 LB LOAD	1.75 IN. PER SEC	0.45	NOT APPLICABLE	ONCE PER FLT.	1.75 IN.
DOOR ACTUATOR-ROTARY	4400 IN-LB	15^{o} PER SEC	0.17	NOT APPLICABLE	ONCE PER FLT.	90^{o}
CANOPY ACTUATORS						
ROTARY AFT HINGED	500 FT-LB	13^{o} PER SEC	0.20	NOT APPLICABLE	1 TO 2 CYCLES PER FLT.	45^{o}
LINEAR, AFT HINGED	800 LB LOAD	3 IN. PER SEC	0.36	NOT APPLICABLE	1 TO 2 CYCLES PER FLT.	10.4 IN.

TABLE 4 ACTUAT[illegible] [illegible]ROL CONFIGURATION AND REDUNDANCY REQUIREM[illegible]TS

SURFACE	NO OF SURFACES PER SIDE	HINGE MOMENT		RATE		NO OF ACTUATORS PER SURFACE
		AFTER FIRST FAIL	AFTER SECOND FAIL	AFTER FIRST FAIL	AFTER SECOND FAIL	
CANARD	1	100%	50%	~100%	~100%	3
ELEVON	1	50%	25%	~100%	~100%	2
SPOILERS	2	50%	*	~100%	*	1
LE FLAPS	3	50%	**	~100%	**	1
RUDDER	1	50%	*	~100%	*	2

* LOSE FUNCTION - BLOWDOWN TO AERODYNAMIC NULL

** LOSE FUNCTION - FAIL TO LAST SELECTED POSITION

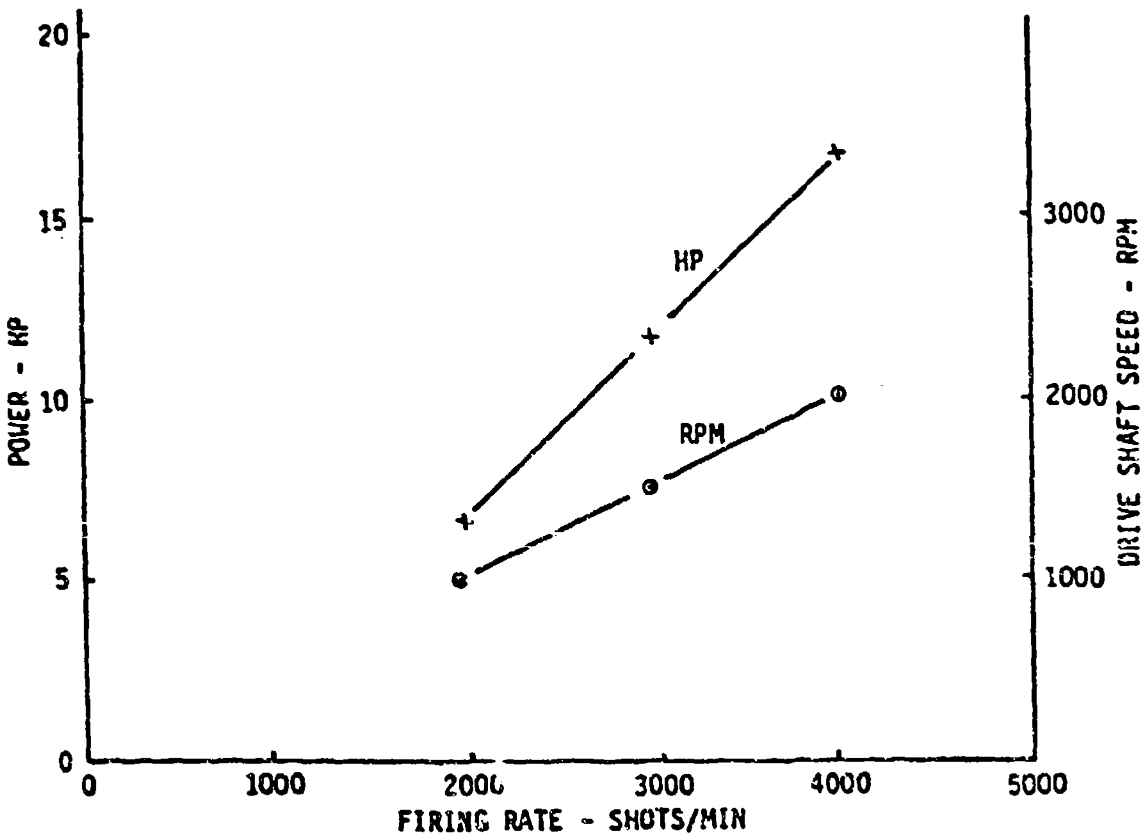


Figure 7 GE 525 Gun Power and Speed vs Firing Rate

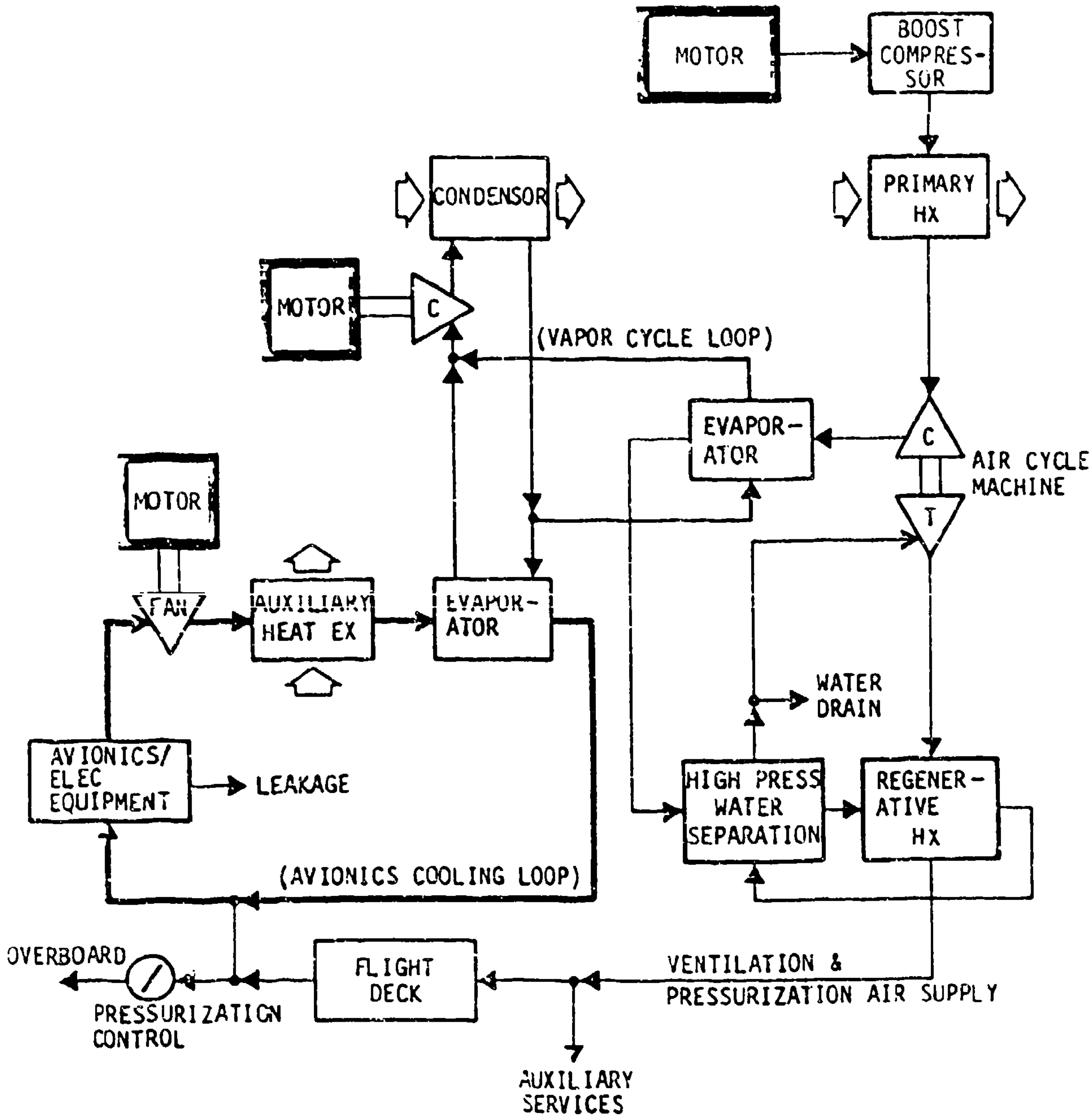


Figure 8 Hybrid Closed-Loop Air-Vapor Cycle ECS

Altitude ft	Airplane Speed	Compressor Pressure Ratio P_R	Corrected Air Flow lbs/min	Compressor Speed rpm
0	Takeoff	1.09	40	15,000
50,000	0.7M	4.27	237	40,000

where corrected flow is defined as

$$\frac{W\sqrt{T}}{p} = \frac{\text{lb/min}\sqrt{°R}}{\text{lb/in}^2}$$

The ECS Compressor motor has to provide 10.7 hp at a fixed speed between 5000 and 22,000 rpm. The ECS fan motor has to provide 42.9 hp at two speeds, 6000 and 12,000 rpm.

2.4 Other Airplane Power Requirements

Power requirements for other air vehicle and avionics subsystems are listed in Table 5. All these requirements are met by electrical power. The kW requirements for these items are the same for the Baseline and for the All-Electric Airplanes, except where noted. The difference is that in the Baseline Airplane these loads are supplied from 400-Hz power whereas in the All-Electric Airplane they are supplied from 270-vdc power. It is assumed that in the 1990 time frame, all these loads will be compatible with either 400-Hz or 270-vdc power.

Loads not listed in Table 5 are the same for either airplane and do not directly impact the trade study. These loads are listed, however, in the detail Baseline Airplane and All-Electric Airplane electrical load analyses. (Sections III and IV)

2.5 Thermal Requirements

A thermal map of the airplane was developed based on aerodynamic heating at Mach 2.2. The skin temperatures are shown in Figure 9. These temperatures

TABLE 5

AIR VEHICLE AND AVIONICS SYSTEM POWER REQUIREMENTS

ITEM	MAX kW LOAD (Total)
Electronics Liquid Cooling Pump*	2.40
Primary Fuel Boost Pump	7.30
Backup Fuel Boost Pump	7.30
Fuel Transfer Pump	7.30
Battery Heater	0.30
Windshield Heater	2.50
Radar (Target Acquisition)	1.50
Weapons Heaters	1.00
Air Data Computer	0.07
Air Data System Heaters	1.50
Integrated Information Management System	5.40
Gun Controls	3.60
Total Temperature Probe Heaters	0.27
JTIDS/TACAN/IFF	0.70
Global Positioning System	0.20
Inertial Reference (Multi-Function)	0.20
Radar (Multi-Function)	5.00
IRCM	2.00
ECM Transmitter	6.00

* All-Electric Airplane only

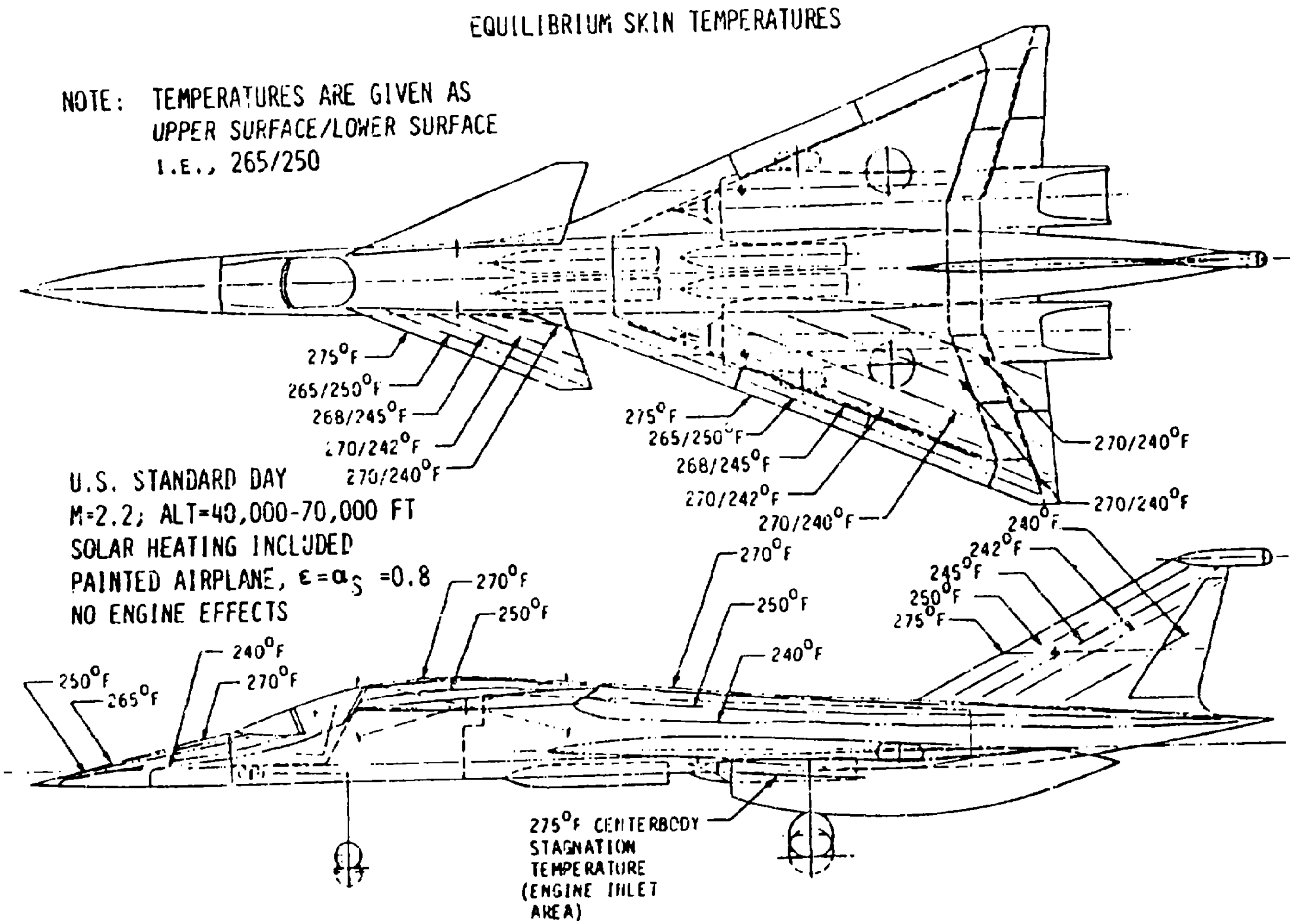


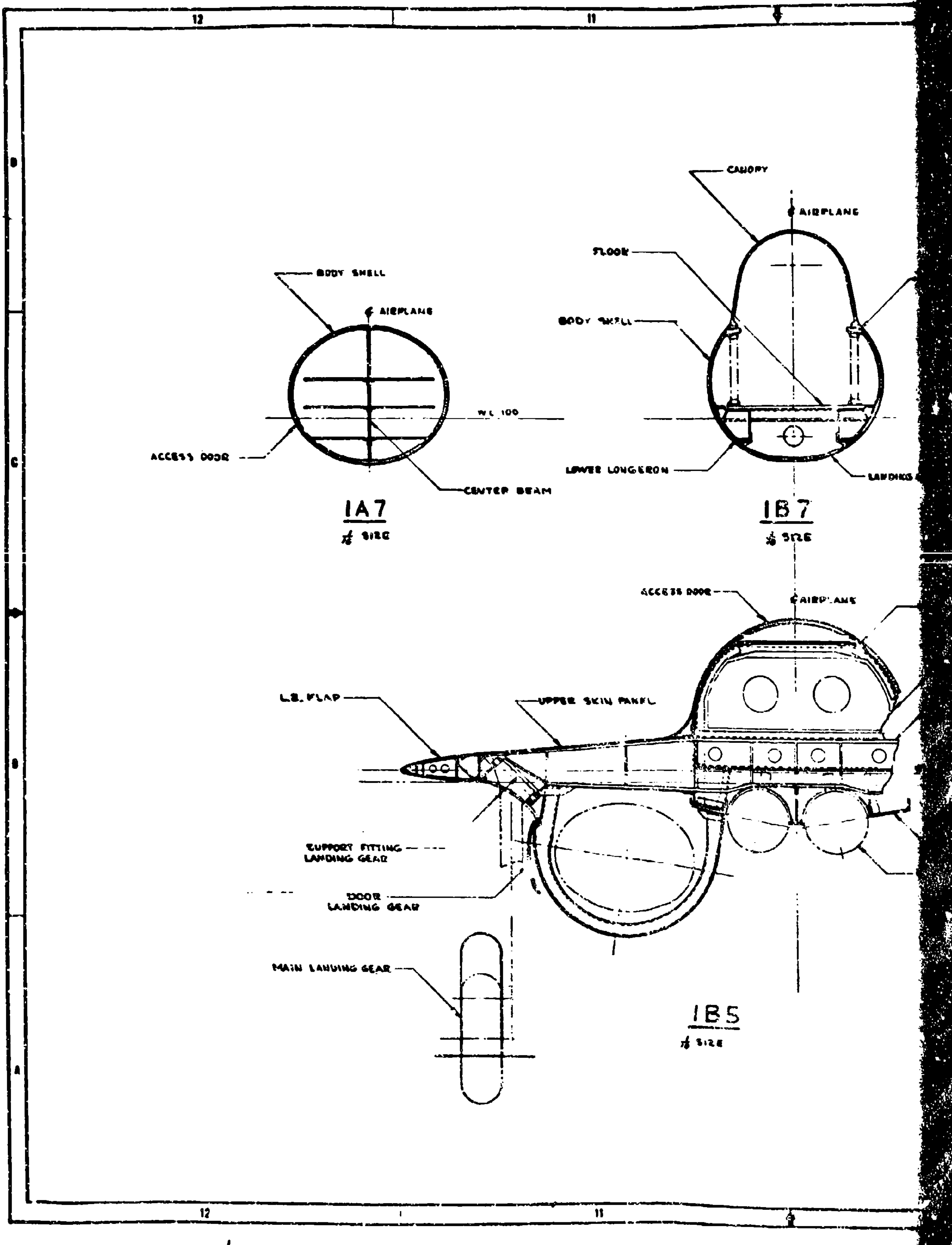
Figure 9 Thermal Map of the Airplane Skin During Supersonic Cruise

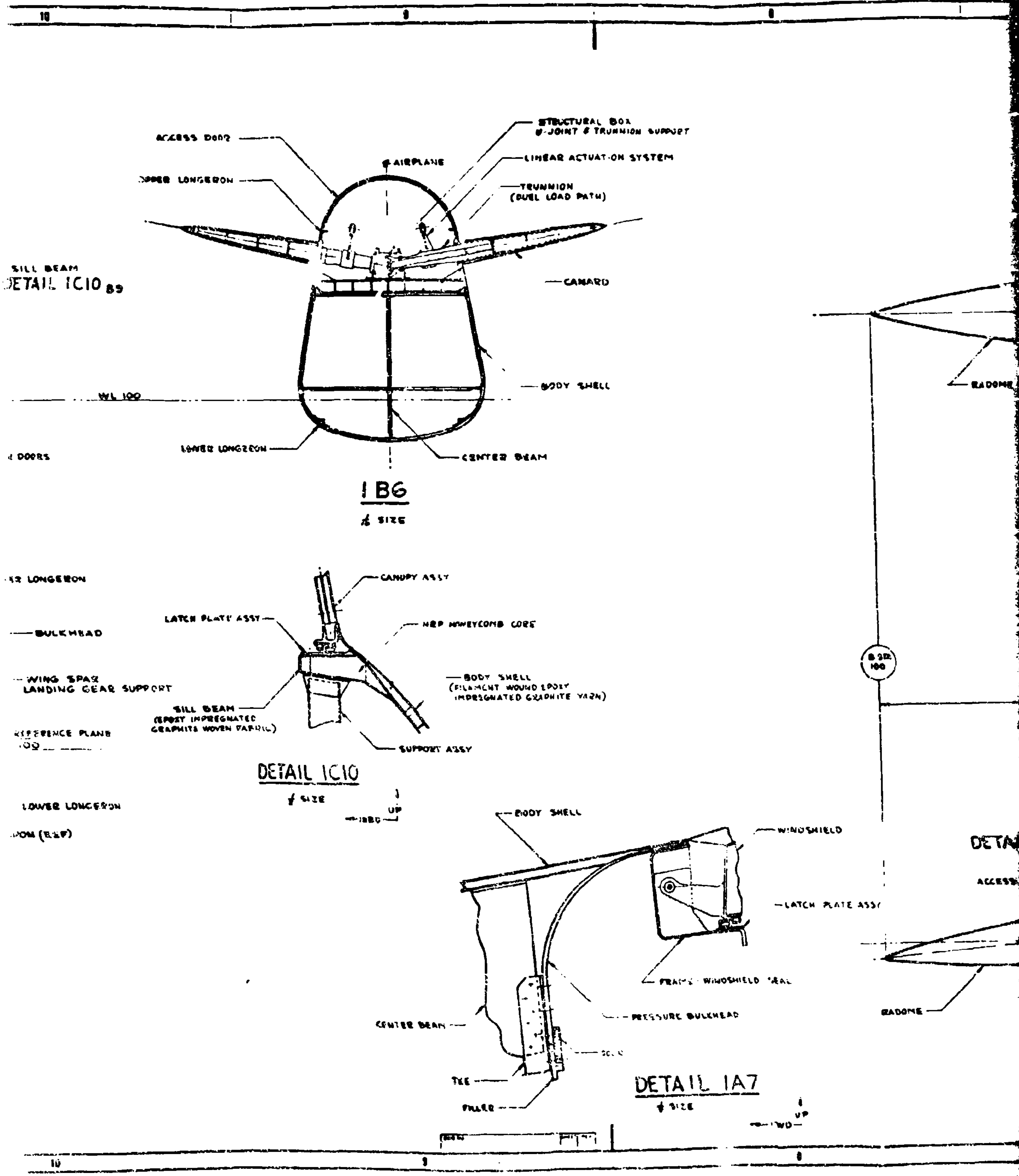
are calculated for a U.S. standard day at Mach 2.2, altitude of 40,000 to 70,000 ft above sea level, include solar heating, and do not include the engine effects.

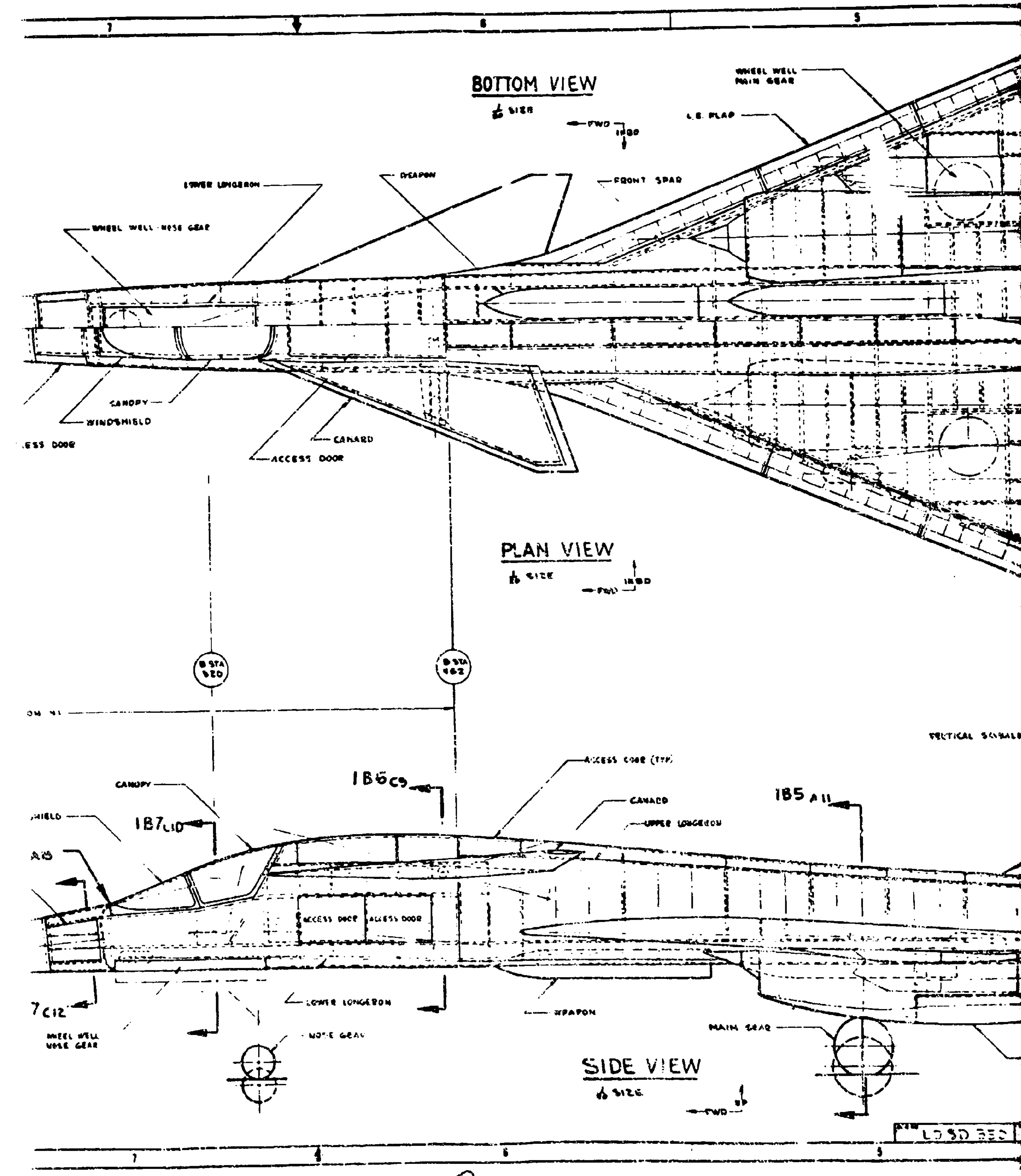
Engine exhaust area temperatures are shown in Figure 6.

2.6 Structural Arrangement

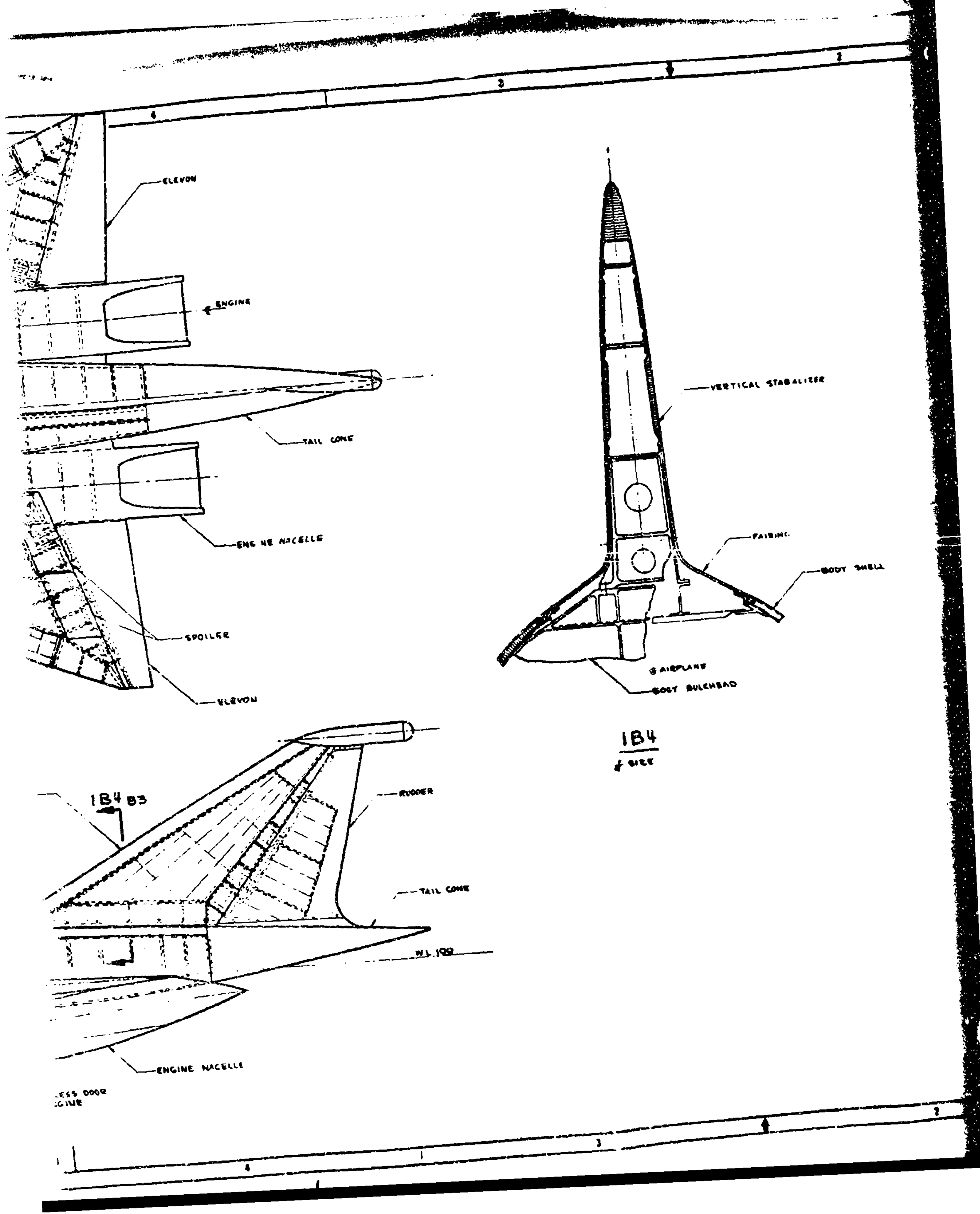
A structural arrangement was also developed for this aircraft and is shown in Figure 10. This was required to determine the exact amount of space available to install the various actuation systems This also facilitated the structural integration of the various actuation system alternatives and selection of the system which would meet this requirement with little or no impact on the aerodynamics of the aircraft.


12
11
CANOPY
¢ AIRPLANE
FLOOR
BODY SHELL
LOWER LONGERON
BODY SHELL
¢ AIRPLANE
W.L. 100
ACCESS DOOR
CENTER BEAM
1A7
1B7
ACCESS DOOR
¢ AIRPLANE
L.B. FLAP
UPPER SKIN PANEL
SUPPORT FITTING
LANDING GEAR
DOOR
LANDING GEAR
MAIN LANDING GEAR
1B5


ACCESS DOOR
STRUCTURAL BOX
U-JOINT & TRUNNION SUPPORT
LINEAR ACTUATION SYSTEM
AIRPLANE
UPPER LONGERON
TRUNNION
(DUEL LOAD PATH)
SILL BEAM
DETAIL 1C10
CANARD
RADOME
BODY SHELL
WL 100
LOWER LONGERON
CENTER BEAM
DOORS
1 B6
SIZE
LONGERON
CANOPY ASSY
LATCH PLATE ASSY
HRP HONEYCOMB CORE
BULKHEAD
WING SPAR
LANDING GEAR SUPPORT
BODY SHELL
(FILAMENT WOUND EPOXY
IMPREGNATED GRAPHITE YARN)
B.SD.
100
SILL BEAM
(EPOXY IMPREGNATED
GRAPHITE WOVEN FABRIC)
REFERENCE PLANE
100
SUPPORT ASSY
DETAIL 1C10
SIZE
UP
LOWER LONGERON
BODY SHELL
WINDSHIELD
DETA
ACCESS
LATCH PLATE ASSY
FRAME WINDSHIELD SEAL
PRESSURE BULKHEAD
RADOME
CENTER BEAM
TEE
FILLER
DETAIL 1A7
SIZE
UP
WD
10
9
8



BOTTOM VIEW
WHEEL WELL MAIN GEAR
L.E. FLAP
FWD
FRONT SPAR
WEAPON
LOWER LONGERON
WHEEL WELL NOSE GEAR
CANOPY
WINDSHIELD
CANARD
ACCESS DOOR
PLAN VIEW
FWD
186 C9
185 A11
187 C10
ACCESS DOOR (TYP)
CANARD
UPPER LONGERON
ACCESS DOOR
ACCESS DOOR
LOWER LONGERON
WEAPON
MAIN GEAR
WHEEL WELL NOSE GEAR
NOSE GEAR
SIDE VIEW
FWD


ELEVON
ENGINE
TAIL CONE
ENGINE NACELLE
SPOILER
ELEVON
VERTICAL STABALIZER
FAIRING
BODY SHELL
℄ AIRPLANE
BODY BULKHEAD
1B4
¼ SIZE
1B4 B3
RUDDER
TAIL CONE
W.L. 100
ENGINE NACELLE

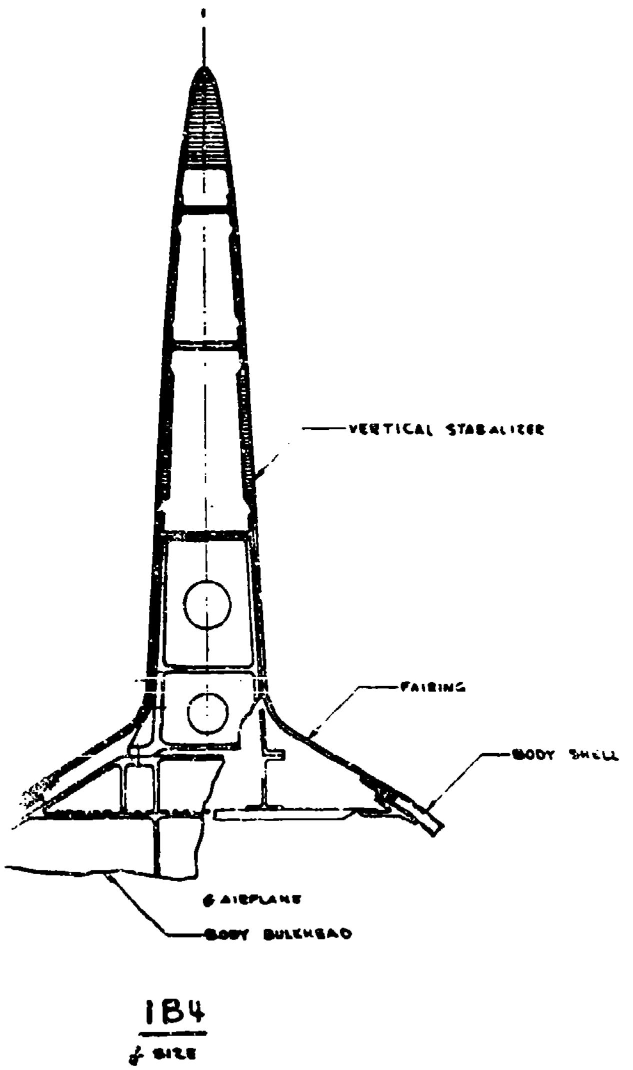


Figure 10 Airplane Structural Arrangement

III BASELINE AIRPLANE CONFIGURATION

3.1 General

The objective of the design phase was to select the most competitive combination of hydraulic actuation systems, hydraulic power systems for transmitting power to those actuation systems, and electrical power systems for providing fly-by-wire control to those actuation systems that could be considered available in the 1990-plus time frame. In keeping with the overall objectives and requirements, it was required that the selected hydraulic power system derive its power primarily from the engine through engine-driven pumps and transmit that power through a distributed system of hydraulic transmission line tubing to the actuation systems. The total secondary power system and the actuation systems are defined so that a direct comparison can be made with the All-Electric Airplane design described in Section IV.

3.2 Actuation Systems for the Baseline Airplane

Consideration was given to various types of power drive units, output mechanisms and control valving arranged in a variety of combinations to suit the particular requirements for the various control functions. The types of power drive units evaluated included piston actuators, vane actuators and multipiston motors. The types of output mechanisms evaluated included bell cranks, rack-and-pinion gearing, helical or ball splines, spur gearing, bent-beam Eccentuators, threaded power screws or ball screws, and planetary or skip-tooth gearing for hinge-line units. The control valve concepts considered were single-stage direct-drive and two-stage electrohydraulic servo valves, staged sequentially-controlled valves, stepper-motor-driven rotary valves, and solenoid valves.

After evaluation of the various actuation systems available, a final configuration was selected for each application. Table 6 summarizes the selected systems for the airplane flight controls and Tables 7 and 8 for the non-flight control functions. Figure 11 shows the location of the actuators in the aircraft and Figure 12 shows how these actuators are integrated into the aircraft structure. Each of the individual applications is covered in the following paragraphs.

TABLE 6 BASELINE AIRPLANE ACTUATION SUMMARY - FLIGHT CONTROLS

Actuator Function	Actuator Type	Piston Area(s)	Stroke Or Deflection	Hyd. Motor
Canard	Linear	7.2 Sq. in. & 3.5 Sq. in.	4.8 inches	---
Elevon	Linear	6.8 Sq. in. & 3.2 Sq. in.	6.7 inches	---
Rudder	Rotary	---	$\pm 30°$	0.313 cipr
Spoiler	Linear	3 Sq. in. & 2.7 Sq. in.	3.6 inches	---
LE Flaps	Linear	7.4 Sq. in. & 1. Sq. in.	1.85 inches	---
Engine Inlet Centerbody	Linear	5.6 Sq. in. & 1. Sq. in.	2.6 inches	---
Engine Inlet Bypass Doors	Rotary	---	90°	0.023 cipr

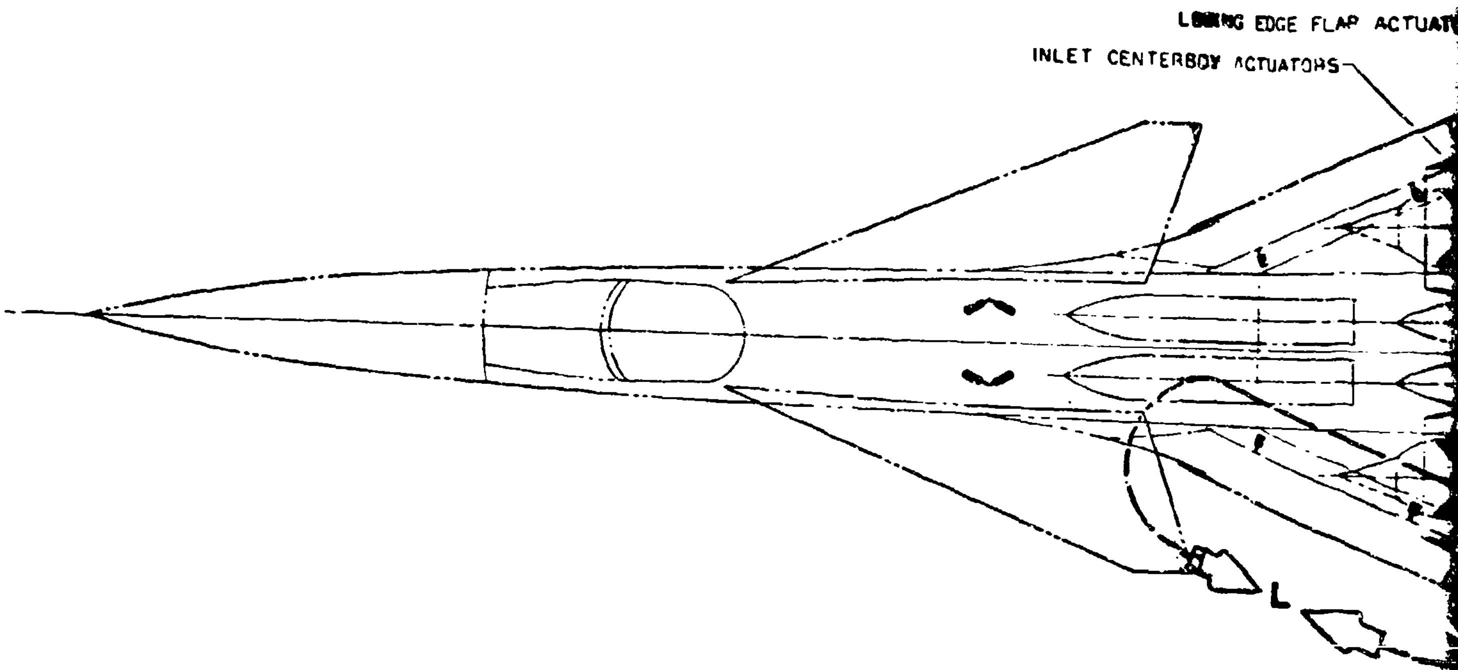
TABLE 7 BASELINE AIRPLANE ACTUATION SUMMARY - LANDING GEAR

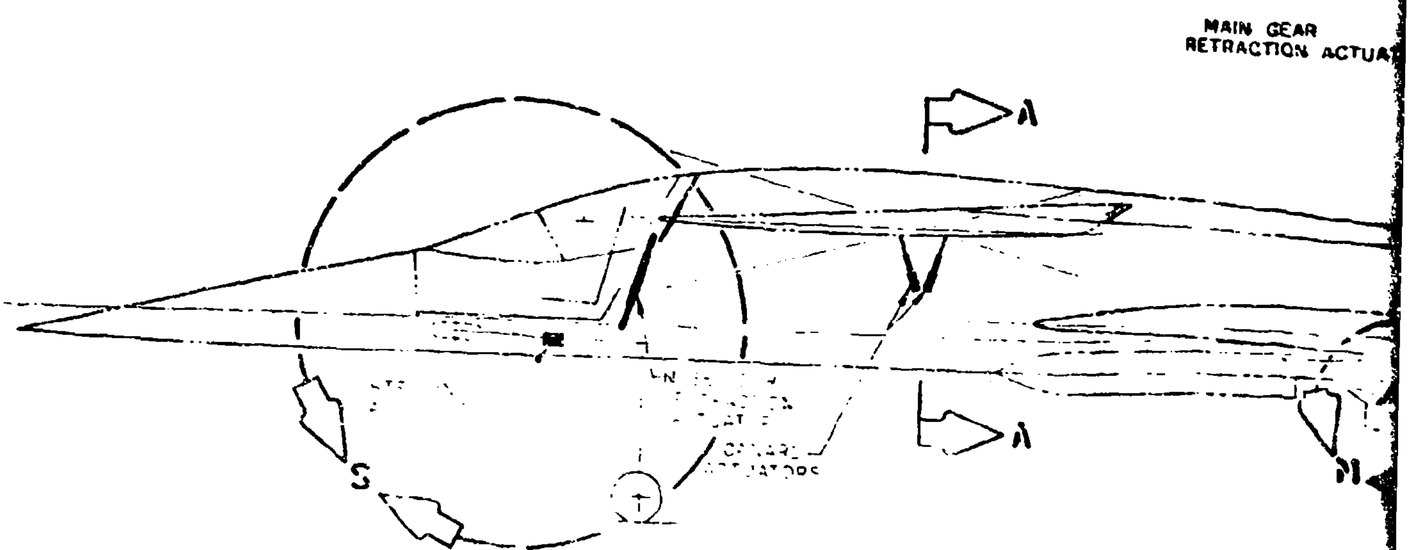
Actuator Function	Actuator Type	Bore Dia.	Stroke Or Deflection
Main Gear Retraction	Linear	3 inches	17 inches
Nose Gear Retraction	Linear	3 inches	14.6 inches
Nose Gear Steering	Rotary	---	$\pm 102°$
Main Gear Brakes	Linear (Multiple Pistons)	---	---

TABLE 8 BASELINE AIRPLANE ACTUATION SUMMARY - MISCELLANEOUS FUNCTIONS

Actuator Function	Actuator Type	Bore Dia.	Stroke	Hyd. Motor
Aerial Refueling Door Actuator	Linear	*	*	-
Aerial Refueling Nozzle Latch Actuator	Linear	*	*	-
Canopy Actuator	Linear	0.925 inch	8.4 inches	-
Gun Drive	Hyd. Motor	-	-	0.34 cipr
ECS Boost Compressor	Driven By AMAD Gearbox	-	-	-
ECS Pack Compressor	Hyd. Motor	-	-	0.10 cipr
ECS Fan	Hyd. Motor	-	-	0.525 cipr
Emergency Generator	Hyd. Motor	-	-	0.375 cipr

* Per existing actuators in Universal Aerial Refueling Receptacle Slipway Installation (UARRSI)


EDGE FLAP ACTUAT
INLET CENTERBODY ACTUATORS
L


MAIN GEAR
RETRACTION ACTUA
A
A
S
M

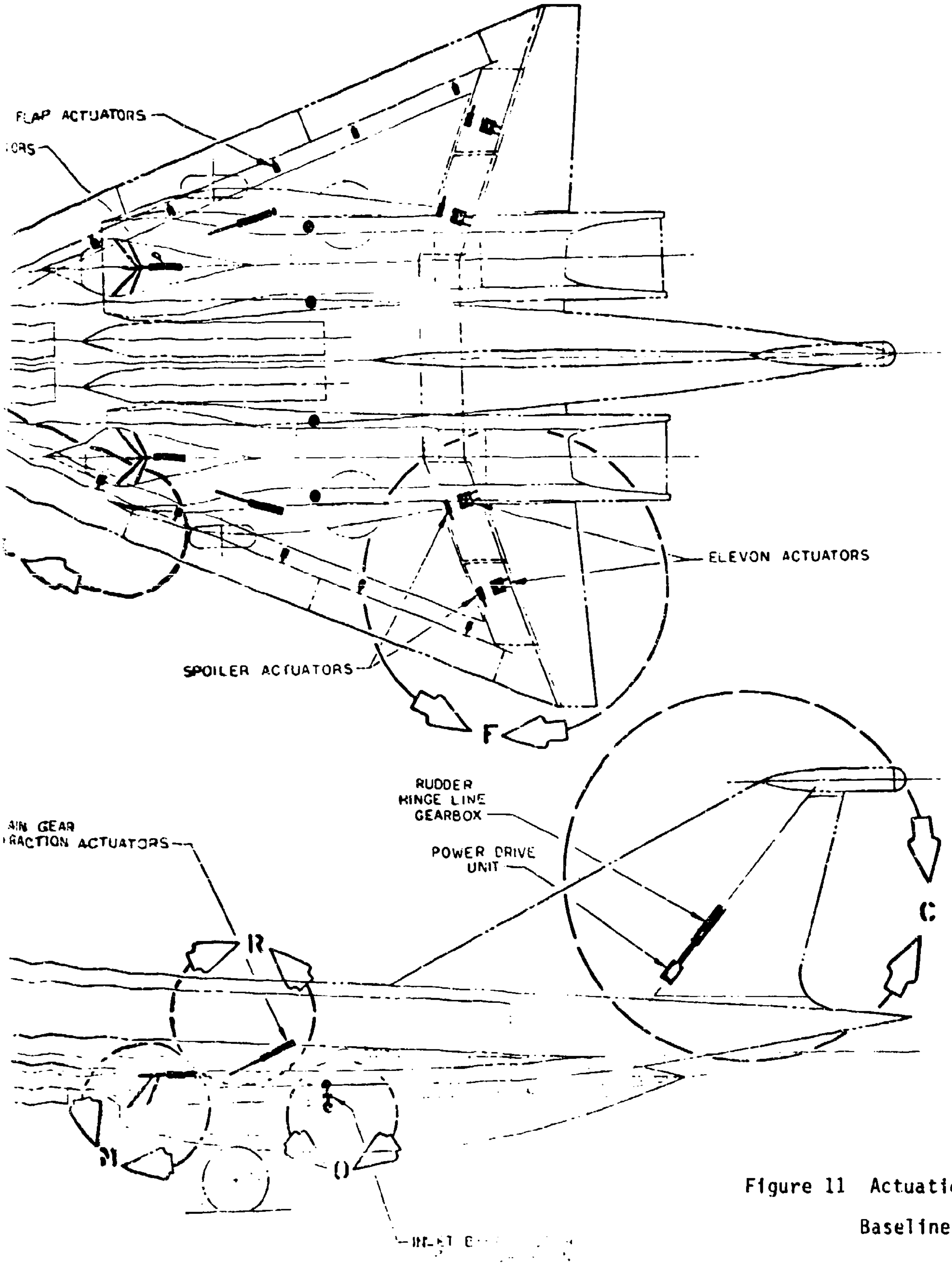
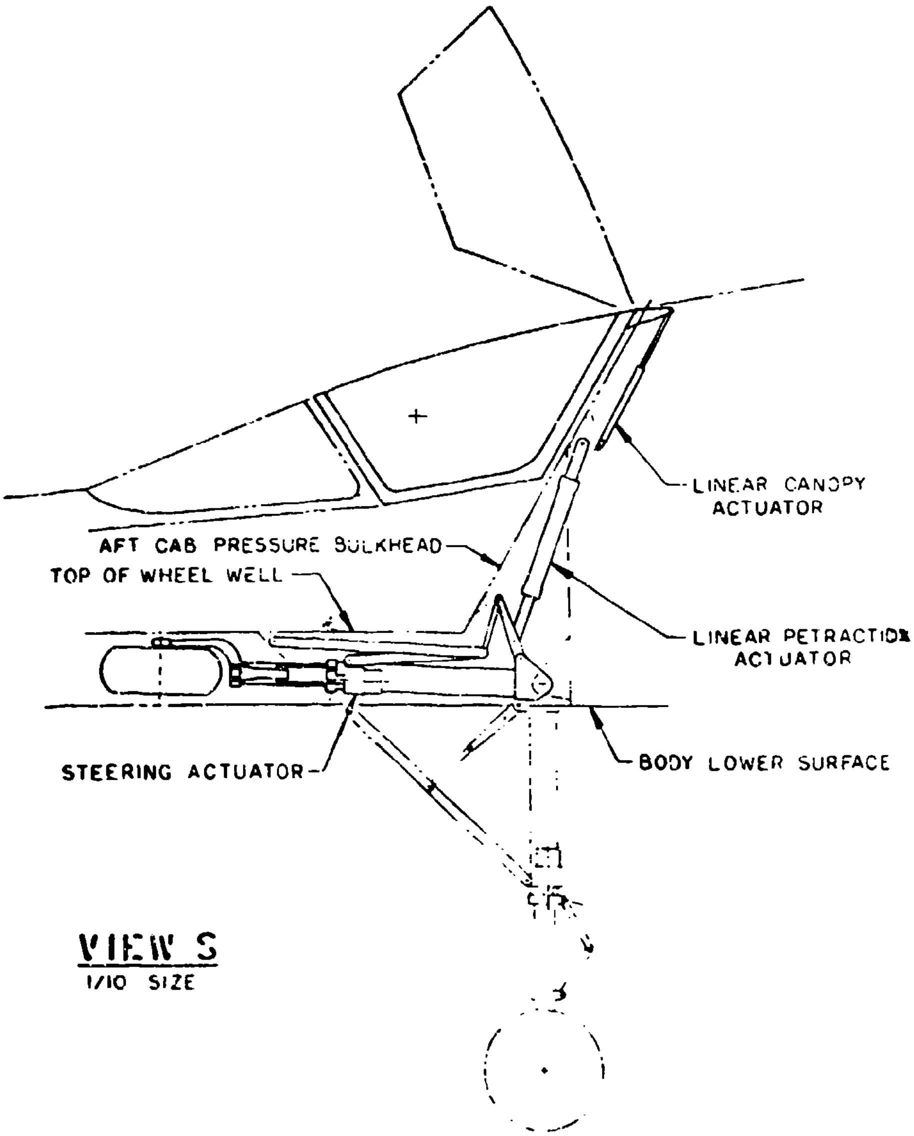
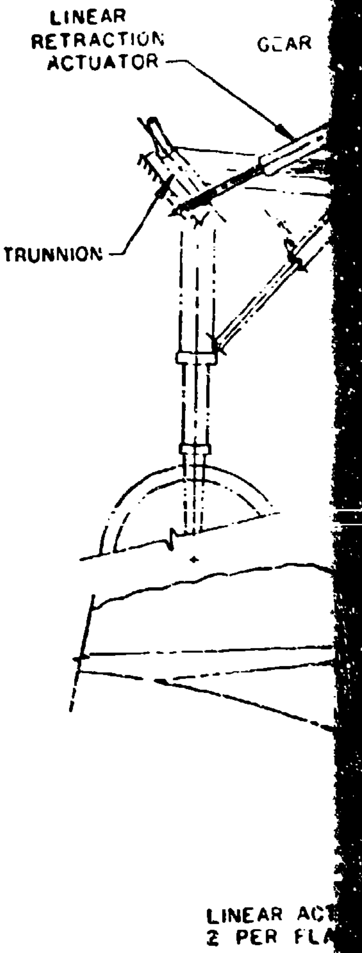


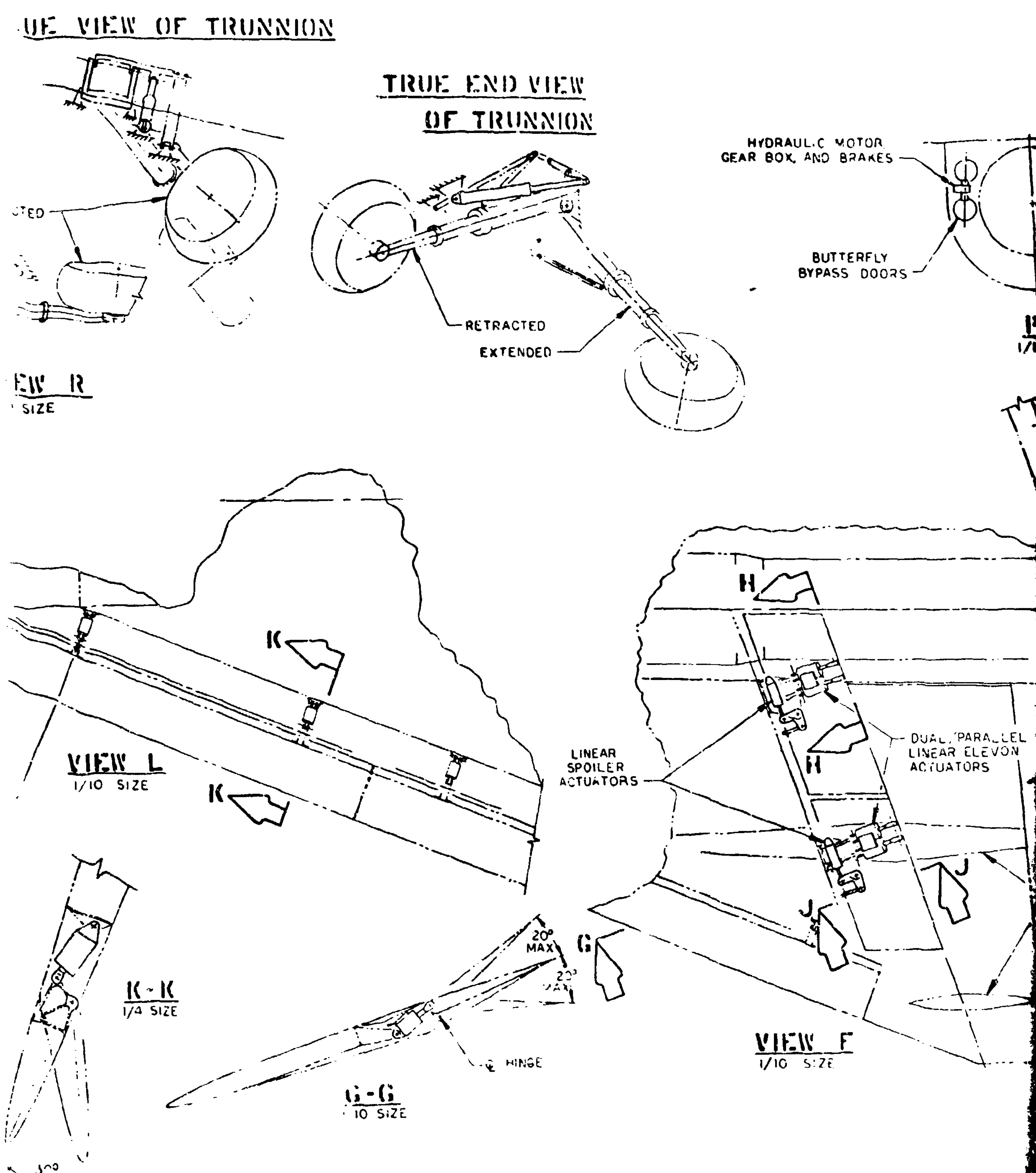
Figure 11 Actuation Systems Location

Baseline Airplane

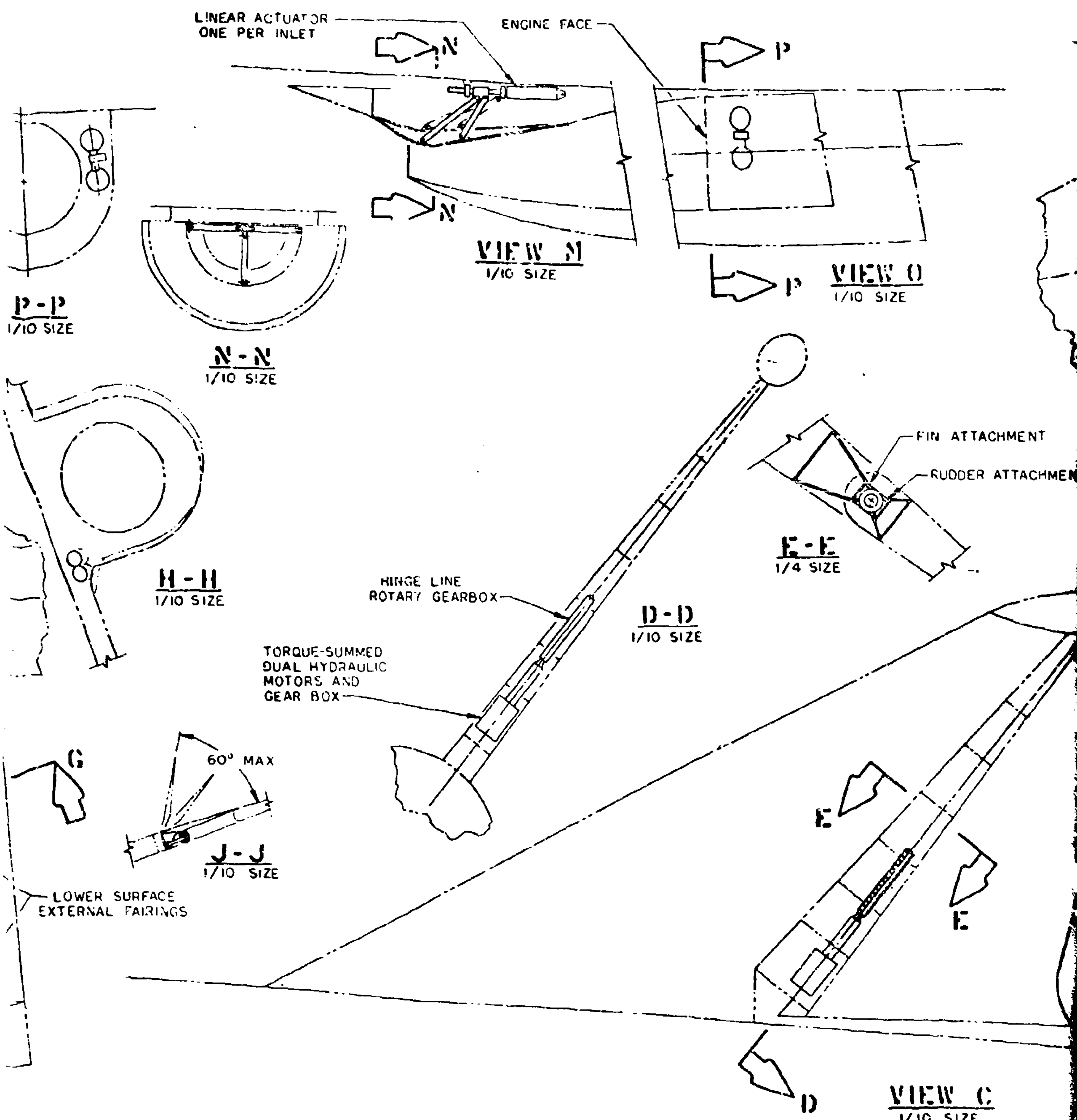


VIEW S
1/10 SIZE





DE VIEW OF TRUNNION
TRUE END VIEW
OF TRUNNION
HYDRAULIC MOTOR
GEAR BOX, AND BRAKES
BUTTERFLY
BYPASS DOORS
RETRACTED
EXTENDED
EW R
SIZE
K
K
VIEW L
1/10 SIZE
LINEAR
SPOILER
ACTUATORS
H
H
DUAL PARALLEL
LINEAR ELEVON
ACTUATORS
J
J
G
20°
MAX
K-K
1/4 SIZE
HINGE
G-G
10 SIZE
VIEW F
1/10 SIZE



LINEAR ACTUATOR ONE PER INLET
ENGINE FACE
N
N
P
P
VIEW M
1/10 SIZE
VIEW O
1/10 SIZE
P - P
1/10 SIZE
N - N
1/10 SIZE
FIN ATTACHMENT
RUDDER ATTACHMEN
E - E
1/4 SIZE
H - H
1/10 SIZE
HINGE LINE
ROTARY GEARBOX
D - D
1/10 SIZE
TORQUE-SUMMED
DUAL HYDRAULIC
MOTORS AND
GEAR BOX
G
60° MAX
J - J
1/10 SIZE
E
E
LOWER SURFACE
EXTERNAL FAIRINGS
D
VIEW C
1/10 SIZE

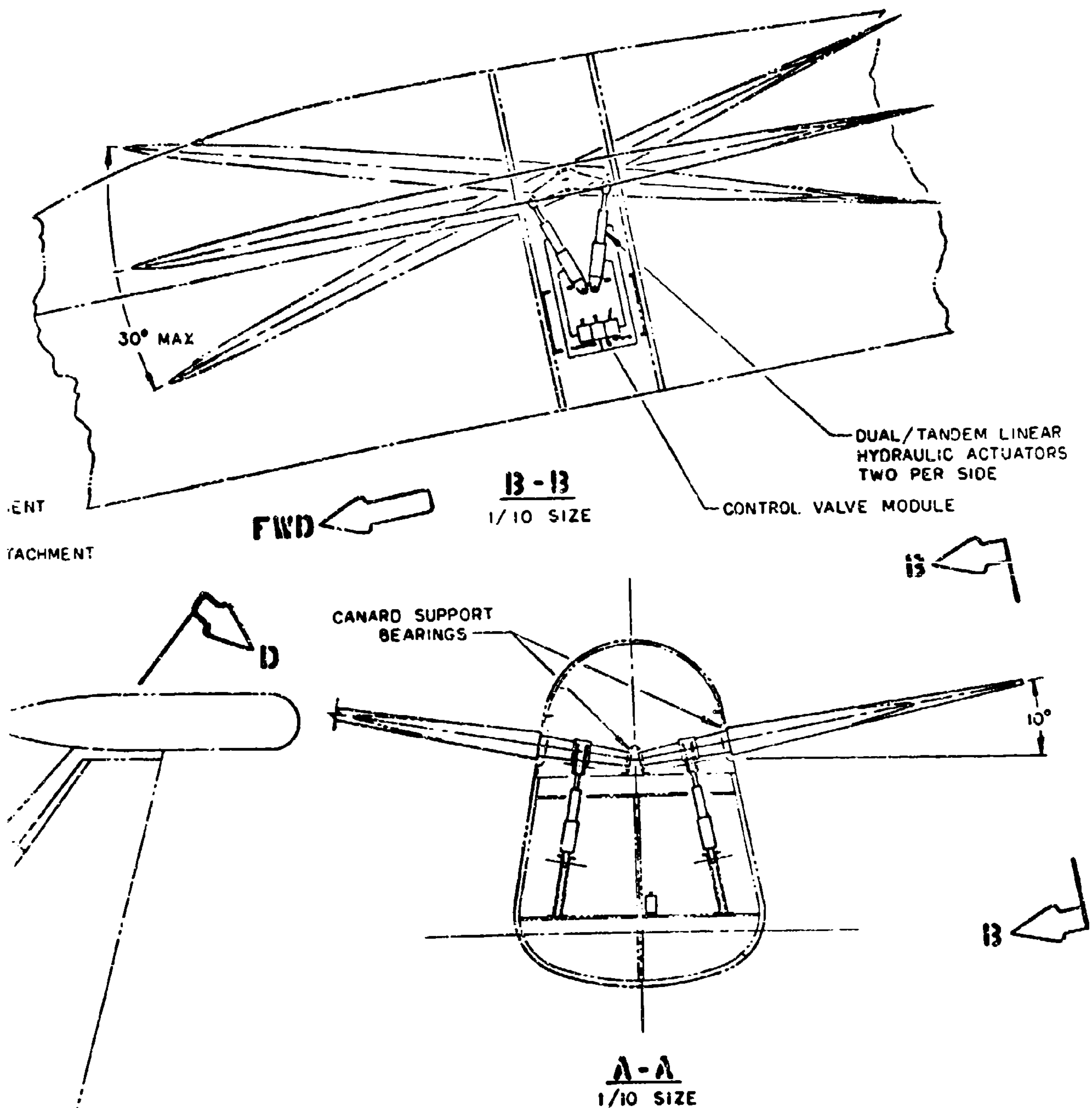


Figure 12 Actuation Systems Installation -

Baseline Airplane

3.3 Flight Control Actuation

3.3.1 Canard

The canard is a critical flight control surface whose continued control is essential for mission completion and safety of flight. Actuation trades considered the two canard surfaces interconnected as well as separated, even though no differential surface control is required since the canard is used only for pitch control. In addition, both linear and rotary actuator designs were evaluated. The selected configuration uses linear actuators independently controlling each canard surface as shown in Figure 12. The following reasons are the basis for this selection:

1. The linear actuator system is lighter. This is because the length of the linear actuator is proportional to the total control surface deflections and the rotary actuator is independent of the control surface deflection. With only 30 degree total surface deflection, linear actuator stroke is only 4.8 inches.
2. Due to the inefficiency of a hydraulic motor and gearbox, the total power consumption of the rotary actuation system would be higher. In addition, a hydraulic motor has a higher internal leakage than the linear actuator. Canards are used for longitudinal trim; and, the steady state aerodynamic load causes more fluid leakage across the hydraulic motors than the linear actuators. This, together with the high duty cycle of the canard surfaces, results in a higher total power consumption.
3. The configuration with no interconnection between the two canard surfaces results in less weight and reduces complexity. The added actuation weight for separate surface control is more than offset by deletion of the interconnecting mechanism and since no additional control capability is needed in terms of increased power, there is no impact on secondary power requirements.

The canard actuation system utilizes four dual-tandem actuators arranged and powered from the three hydraulic systems to meet the redundancy requirement as specified in Table 4. Tandem actuators are used because they can be placed close to the surface to maintain adequate stiffness between the actuator rod and the canard surface.

Each dual-tandem actuator consists of a full-area piston and a half-area piston. Any two of the three hydraulic systems can drive both canard surfaces at 100% of the design hinge moment; 50% from system #1 through the two forward actuator full-area pistons, 50% from system #2 through the two aft actuator full-area pistons, and 50% from system #3 through all four actuator half-area pistons. Under normal conditions, (all 3 hydraulic systems operating) each tandem actuator is capable of providing 75% of the surface design hinge moment.

Valves are sized to meet the rate requirement at maximum load. A flow limiter, limiting the maximum rate to 70 degrees per second, avoids excessive flow at the no-load condition.

Actuation system components for each of the two canard surfaces consists of the following:

Dual-tandem linear actuator (2 required @ 39 pounds each)	78.0 pounds
Control Valve Module	7.0 pounds
Total Weight, per surface	85.0 pounds

3.3.2 Elevons

The elevon control surfaces have a dual role to provide both longitudinal and lateral control of the airplane. Actuation trades considered both linear and rotary actuator designs as well as installation of part of the system in the body. The hinge moment requirements for the elevons are large and the available space for equipment installation is small due to the thin wing geometry. Configuration studies indicated that both linear and rotary actuation equipment exceeded the designated envelope.

Since the maximum hinge moment when moving the trailing edge down is roughly twice as large as the maximum hinge moment when moving it up, an unequal-area linear actuator can be used with the piston head-end area sized to meet the larger load and the rod-end area sized to meet the smaller load, whereas the rotary actuator has to be sized to meet the larger load. The linear actuator is the more efficient approach due to the inefficiency of a hydraulic motor/gearbox arrangement. Also, since the elevon surfaces are used for longitudinal trim, the steady-state aerodynamic loads would cause more fluid leakage across the hydraulic motors than the linear actuators.

Therefore, the choice of the linear actuator for the elevon function results in a lighter system with less power consumption. Consideration was given to installing the actuators in the body to avoid exceeding the envelope requirement. However, the torque tubes required to carry the load to the elevon became unreasonably large and heavy. A detailed study of the airplane structure and geometry determined that an increased number of smaller diameter linear actuators with shorter moment arms could be used to better fit the envelope with less fairing.

The selected configuration (Figure 12, View F) uses four actuators (two dual/parallel linear actuators) per surface to meet the hinge moment requirements with minimum actuator dimensions and fairing. Each of the four actuators weighs 75 pounds.

The increase in drag due to the elevon actuator fairing on the baseline airplane is two-tenths of one percent of the total airplane cruise drag. The resulting impact on specific fuel consumption will be negligible and no further consideration will be given to this subject in the trade study.

The actuator and valve are sized to meet the rate requirement at maximum load and also meet the maximum rate of 70 degrees/sec at no load. No flow limiters are used. The major actuation characteristics are:

Actuator piston area	6.8 in^2 head end, 3.2 in^2 rod end
Moment Arm	10 inches
Stroke (Total)	6.7 inches

3.3.3 Rudder

The rudder control surface provides directional control of the airplane. Actuation trades considered both linear and rotary actuation.

The rotary actuation system, Figure 12, View C, was chosen for the rudder function for the following reasons:

(1) Envelope restrictions require that linear actuators be placed in the aircraft body which in turn requires a long torque tube to carry the load evenly to the surface. Also, the large surface deflection, 60 degree total, requires a relatively long linear actuator. These two factors result in a greater weight for the linear actuation system. The rotary actuation system is able to fit in the designated envelope and is able to handle the large surface deflection with less weight.

(2) Due to the inefficiency of the hydraulic-motor/gearbox, fluid leakage and peak power consumption of the rotary actuation system is higher. However, the rudder load and duty cycle are relatively low and power consumption caused by internal fluid leakage across the hydraulic motor is low.

One configuration considered used three hinge-line gearboxes to distribute the load to the rudder surface. However, after detailed study of the structure, geometry, and gearbox design, it was determined a single hinge-line gearbox was more desirable and would result in a weight saving.

The selected system consists of a power drive unit, including two hydraulic motors, control valves and a torque-summed reducing gearbox installed in the body. A torque tube is used to carry the load to the single hinge-line gearbox attached to the surface. Hydraulic motors are sized to meet the rate requirement at maximum load. No flow limiter is required.

The actuation system for the rudder consists of the following components:

Hydraulic Motor (2 required @ 7.5 lbs)		15.0 pounds
Hingeline Gearbox		22.0 pounds
Reduction Gearbox		11.0 pounds
	Total Weight	48.0 pounds

3.3.4 Spoilers

The spoiler control surfaces provide, in conjunction with the elevons, lateral control of the airplane. Actuation trades considered both linear and rotary actuation.

Selection of a linear actuation system instead of a rotary actuation arrangement was influenced by the following:

(1) An unequal-area linear actuator to handle unequal loads results in a lighter system and lower power consumption than a rotary actuation system.

(2) Spoilers are fairly inactive during normal flight. The surfaces are retracted most of the time and the actuators or the motors are positioned to hold against the upward aerodynamic load. The hydraulic motor in a rotary actuation system with larger internal fluid leakage consumes more power due to holding this load. A hydraulic check valve is usually provided in the hydraulic supply line of the linear actuator to prevent back driving when the aerodynamic load exceeds the actuator capability. Use of the check valve is not effective in the rotary actuation system because of the higher internal leakage across the motor.

The selected system, Figure 12 View F, consists of an unequal-area linear actuator driving each of the four spoiler segments. Each actuator weighs 17.8 pounds.

The larger actuator area (piston end) is active when the actuator is holding the spoiler trailing edge down, while the larger area (rod end) is active when the actuator is forcing the trailing edge up. A flow limiter is used to reduce excessive flow in the no-load condition.

3.3.5 Leading Edge Flaps

The original linear actuator design approach was to tie all leading edge flap surfaces together and actuate by two linear actuators installed in the body.

This was found impractical due to the large torque tube required to carry the load out to the flaps. The alternative, shown in Figure 12 View L, uses two linear actuators, powered by a single hydraulic system, to control each flap segment and is the approach selected for the Baseline Airplane. Since the aerodynamic load is only exerted in one direction, an unequal-area actuator is used. A blocking valve and bypass valve are required so that the actuator will remain in the last selected position in the event of total power loss. A flow limiter is required to limit the actuator rate in the no-load condition. A total of 12 actuators are required, each with a weight of 19.3 pounds.

A rotary actuation scheme, consisting of a body-mounted power drive unit driving through a torque tube and angle gearbox to hingeline gearboxes, was also considered. The rotary actuation approach and the original linear approach, with all leading edge flap segments connected together, were abandoned in favor of the selected approach because:

(1) Total surface deflection is small and aerodynamic load is only in one direction.
(2) Because of the inefficiency of the gearboxes and hydraulic motors, the rotary configuration is heavier and consumes more power. The flaps are required to operate during descent and landing when the hydraulic power supply is low due to lower engine power settings.
(3) With all flaps tied together, there is a remote chance for asymmetric deployment in the event of a structural failure. Each linear actuator incorporates a blocking valve so that in case of failure, such as loss of hydraulic power, the flap will remain in the last selected position. Structural damage, or both actuators leaking, could cause one flap to blow back which is less serious (and is considered acceptable) than all three flaps failing together.

3.4 Engine Inlet Control Actuation

3.4.1 Engine Inlet Centerbody

The function of this actuation system is to drive a linkage assembly that moves the inlet centerbody ramp which in turn expands or contracts the centerbody radially thereby regulating the speed of the incoming air.

Both linear and rotary actuation schemes were considered. Since the aerodynamic load is in one direction only, an unequal-area linear actuator proves to be considerably lighter than the less efficient rotary actuation system.

The general arrangement is shown in Figure 12 View M. The actuator and valve are sized to meet the maximum rate at maximum load. A flow limiter is used to limit flow in the no-load condition. One actuator is required per engine, with a weight of 18.0 pounds each.

3.4.2 Engine Inlet Bypass Doors

As shown in Figure 12 View P-P, there are four bypass doors for each engine. The aerodynamic loads are small but the doors are required to open up to 90 degrees.

Both rotary and linear actuation systems were considered for this function with the choice going to the rotary system for the following reasons:

(1) A rotary system is more suited to large deflection angles; a linear actuator would experience nonlinear motion at large deflection angles.

(2) A rotary system is more compact for this application.

The actuation system for each of the 4 pairs of bypass doors consists of the following components:

Rotary Vane Actuator	4.0 pounds
Total Weight per pair of doors	4.0 pounds

3.5 Landing Gear and Brakes

The hydraulic actuation concepts traditionally used for landing gear retraction, steering, and brakes, and for the other utility subsystems, have been highly refined over the past 40 years. Except for the few exceptions noted, no improvement could be found in deviating from the normal practice other than using the increased pressure level selected for this ATS study aircraft (See Section 3.10.3). For landing gear retraction, unbalanced-piston actuating cylinders operating through appropriate bellcranks generate the required force moment to lift the gear against its combined dead weight and aerodynamic loads. With built-in snubbing provisions, they can cushion the load at either end of the stroke including the bottoming load due to emergency free-fall extension. All components are covered in the following paragraphs except the isolation valves (2 at 2.0 pounds each), and the 3-position control valve (1 at 3.0 pounds).

3.5.1 Main Gear Retraction

The retraction/extension system for the main landing gear consists of two linear piston actuators, one for each main gear, controlled by one solenoid valve. Landing gear doors are slaved to the gear strut, and uplocks and downlocks function through the motion of the actuator and mechanical linkage. This is an improvement over some existing aircraft which require separate actuators for actuating doors and position locks. In addition, like most aircraft, the system allows emergency free-fall extension following manual release of the uplock by the pilot. The installation is shown in Figure 12 View R.

The selected actuator extends during gear retraction and retracts during gear extension with snubbing provided at the retracted (gear extended) end. The actuator weight for each of the two main gears is 18.9 pounds.

3.5.2 Nose Gear Retraction

The retraction/extension system for the nose gear consists of one linear piston actuator in a system similar to that described for each main gear. The actuator is controlled by the same solenoid valve used for the main gear. The installation is shown in Figure 12 View S.

The selected actuator retracts during gear retraction and extends during gear extension. Actuator weight is 29.5 pounds.

3.5.3 Nose Gear Steering

Nose gear steering is provided by an actuator module, consisting of a vane type rotary power drive unit with spur gear output, electrohydraulic position servovalve, and associated functional circuits. It is mounted on the nose gear strut and drives a strut-mounted ring gear as shown in Figure 12 View S. Actuator weight, including the hydraulic motor, is 22 pounds.

3.5.4 Main Gear Wheel Brakes

The main gear wheel brakes are multiple disk type using advanced composite carbon heat sink material. Actuation arrangement is the standard multiple hydraulic pistons in a brake housing sized for 5000-psi operating pressure. Two brakes are required, one per each main wheel.

The brake actuation components have been segregated from the total brake assembly in order to permit a more meaningful comparison with the All-Electric Airplane. The brake actuation system for each of the two main gears consists of the following components:

Piston Actuators (8 required @ 0.5 lb)	4.0 pounds
Wear Adjustors (8 required @ 1.0 lb)	8.0 pounds
Control Valve Module	9.0 pounds
Shutoff Valve	1.0 pound
Parking Valve	2.5 pounds
Accumulator (including 3 pounds fluid)	13.0 pounds
Total, per gear	37.5 pounds

3.6 Aerial Refueling System

A standard universal aerial refueling receptacle slipway installation (UARRSI) is provided. For this study, the current 3,000-psi actuation system with two linear piston actuators, the slipway door actuator and the nozzle latch actuator is used along with a pressure reducing valve to reduce the 5,000-psi system pressure to 3,000 psi for this subsystem. Actuation system weights are:

Refueling Door Actuator	1.5 pounds
Nozzle Latch Actuator	1.0 pound
Control Valve	3.3 pounds
Total	5.8 pounds

3.7 Canopy Actuation

Due to the relatively large overhanging moment, a linear piston actuator with an operating lever arm as shown in Figure 12 View S, was selected. An internal locking mechanism holds the actuator in its retracted (canopy open) position, and internal snubbing is provided at both ends of its stroke. Actuation system weights are:

Linear Actuator	2.9 pounds
Control Valve	1.0 pound
Total	3.9 pounds

3.8 Gun Drive

A hydraulic motor is used to drive the 25-mm Gatling-type gun rotor similar to the currently used 20-mm and 30-mm gun drives. One motor is used to drive the gun barrel and the ammunition feed system which require 14 hp and 11 hp respectively at the design firing rate of 3,600 rounds per minute. For this study, a 0.34 cu. in. per rev. (cipr) motor operating at 7,200 rpm drives the main gun system drive shaft at 1,800 rpm through a 4:1 speed-reducing gearbox. Component weights are as follows:

Gun Drive Gear Box	10.0 pounds
Hydraulic Motor	7.6 pounds
3-Position Control Valve	8.4 pounds
Total	26.0 pounds

3.9 Environmental Control System (ECS)

In order to minimize engine fuel consumption on aircraft in the 1990 time frame, bleed-air extraction as traditionally used for the ECS pack will probably not be permitted. Since the weight and drag penalties for shaft power extraction are considerably lower than for bleed-air extraction, it is assumed that the ECS power unit components must be driven either directly by the engine or by hydraulic or electric motors. The environmental control system has three power drive components as described in the following paragraphs. The system schematic diagram is shown in Figure 13.

3.9.1 ECS Boost Compressor

The ECS boost compressor raises ram air pressure to meet the pressure demands of the ECS pack. It is a continuous-duty unit with a speed range from 15,000 to 40,000 rpm, and a maximum output of 50 hp. The boost compressor is mounted on the right hand engine-driven airframe-mounted accessory-drive (AMAD) gearbox.

3.9.2 ECS Pack Compressor

The ECS pack compressor compresses the working fluid, air or freon, used by the refrigeration pack. It is a continuous-duty unit with a fixed speed between 5,000 and 23,000 rpm and an output power requirement of 10.7 hp. For this study, a 0.10-cipr motor drives the compressor directly at 10,000 rpm. The hydraulic motor and associated 2-position control valve weigh a total of 5.0 pounds.

3.9.3 Electronic Cooling Fan

The electronic cooling fan circulates air between the heat sink, provided by the ECS refrigeration pack, and the electronic equipment. It is a continuous-duty two-speed unit running at 6,000 rpm during subsonic flight and 12,000 rpm during supersonic flight and draws 21.5 and 42.9 hp respectively at those speeds. For this study, a 0.525-cipr motor drives the fan through a 1.5:1 speed-increasing gearbox. Component weights are as follows:

Gear Box	7.5 pounds
Hydraulic Motor	7.6 pounds
3-Position Control Valve	1.0 pound
Total	16.1 pounds

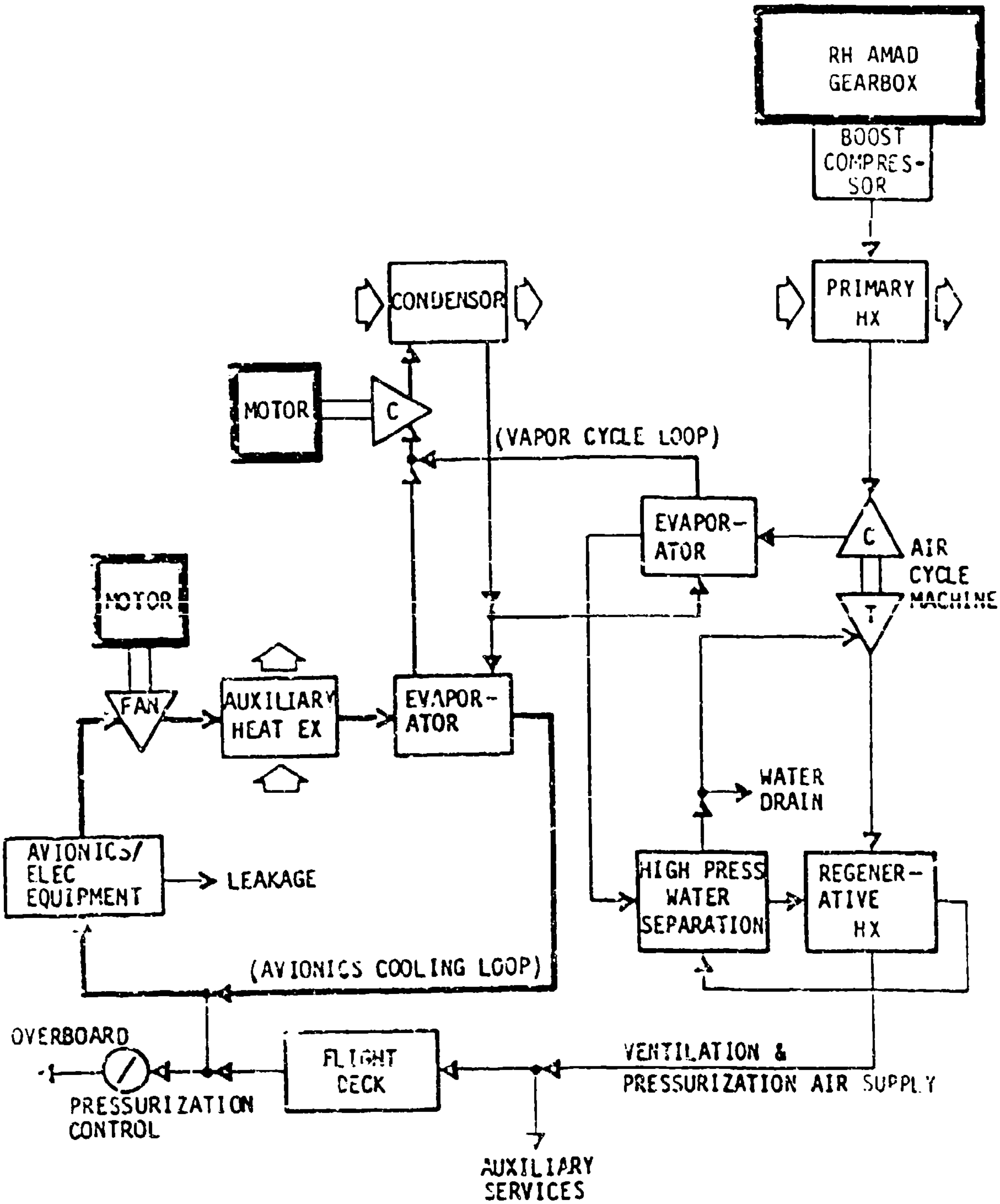


Figure 13 Environmental Control System (ECS) - Baseline Airplane

3.10 Secondary Power System

3.10.1 General Arrangement

During Phase II several Secondary Power System and subsystem arrangements were devised, studied, and evaluated. This and the following sections summarize that effort and describe the selected system.

A significant factor in the development of the secondary power generation system arrangement is the ability to drive the engine-driven hydraulic pumps and electrical generators on the ground for system checkouts without powering the main engines. This led to the selection of airframe-mounted accessory-drive (AMAD) gearboxes which can be declutched from the main engines for the ground checkouts and reclutched for normal operation. Such units were developed for the Boeing supersonic transport and have been used on several recent military aircraft including the B-1 bomber, and the F-15, F-16, and F-18 fighters.

Another significant factor is to provide power for starting the main engines without external power sources. Three types of engine starters were considered: a solid propellant or liquid propellant cartridge unit for each engine which supplies hot gas to an air turbine starter on each engine; a gas turbine APU which provides hot gas to an air turbine starter on each engine; or, a gas turbine APU or jet fuel starter which provides shaft power to each engine.

The last choice was favored since it can also provide shaft power to the AMAD gearboxes to drive the main hydraulic pumps and generators for ground checkouts. Of the several types of gas turbine power units which could be considered, the LOX/JP-4 integrated power unit (IPU) was chosen as the most promising. This concept, which is being developed by the Rocketdyne Division of Rockwell International under Air Force Aero Propulsion Laboratory contract can operate either in a bipropellant power mode, with aircraft fuel (JP-4) and liquid oxygen (LOX) oxidizer, or in a standard gas turbine mode with JP-4 fuel and outside air.

The selected arrangement is shown in Figure 14 and the drive system components and weights listed in Table 9. The LOX/JP-4 IPU and angle gearbox, both normally declutched in flight, are connected to the AMAD gearboxes for ground checkout of the hydraulic and electrical systems and for engine starting. The normal sequence is to start the IPU with the LOX/JP-4 gas generator and then immediately switch to the gas turbine mode in order to conserve LOX. Then, one or both AMAD gearboxes can be connected for system checkouts. The engine power-takeoff shafts can be connected for engine starting following which the IPU can be shut down and the angle gearbox declutched from each AMAD gearbox. Each AMAD gearbox remains connected to its adjacent engine throughout the normal flight operations.

During an emergency situation where either engine suffers a flameout, shaft power can be extracted either from the opposite engine or the IPU for starting the disabled engine and keeping its AMAD gearbox running. In the event of simultaneous loss of power from both engines, the IPU can be started in the LOX/JP-4 mode immediately at any altitude and provide sufficient power to start engines and drive the AMAD gearboxes. If engine starting cannot be accomplished, the IPU continues to drive the pumps and generators on the AMAD gearboxes so that the pilot can maintain vehicle attitude as necessary for an engine start at lower altitude or for a safe ditching or bailout.

3.10.2 Electrical Power System

The electrical power system for the Baseline Airplane is required to provide electrical power in accordance with the requirements of MIL-E-25499 and MIL-STD-704C. It must provide source redundancy for supplying power to the fly-by-wire flight control system and other flight-critical loads in the Baseline Airplane configuration. The electrical power system includes generators, power conversion equipment, distribution circuits, and associated control and protection devices.

Three different electrical power generation concepts were comparatively evaluated during Phase II:

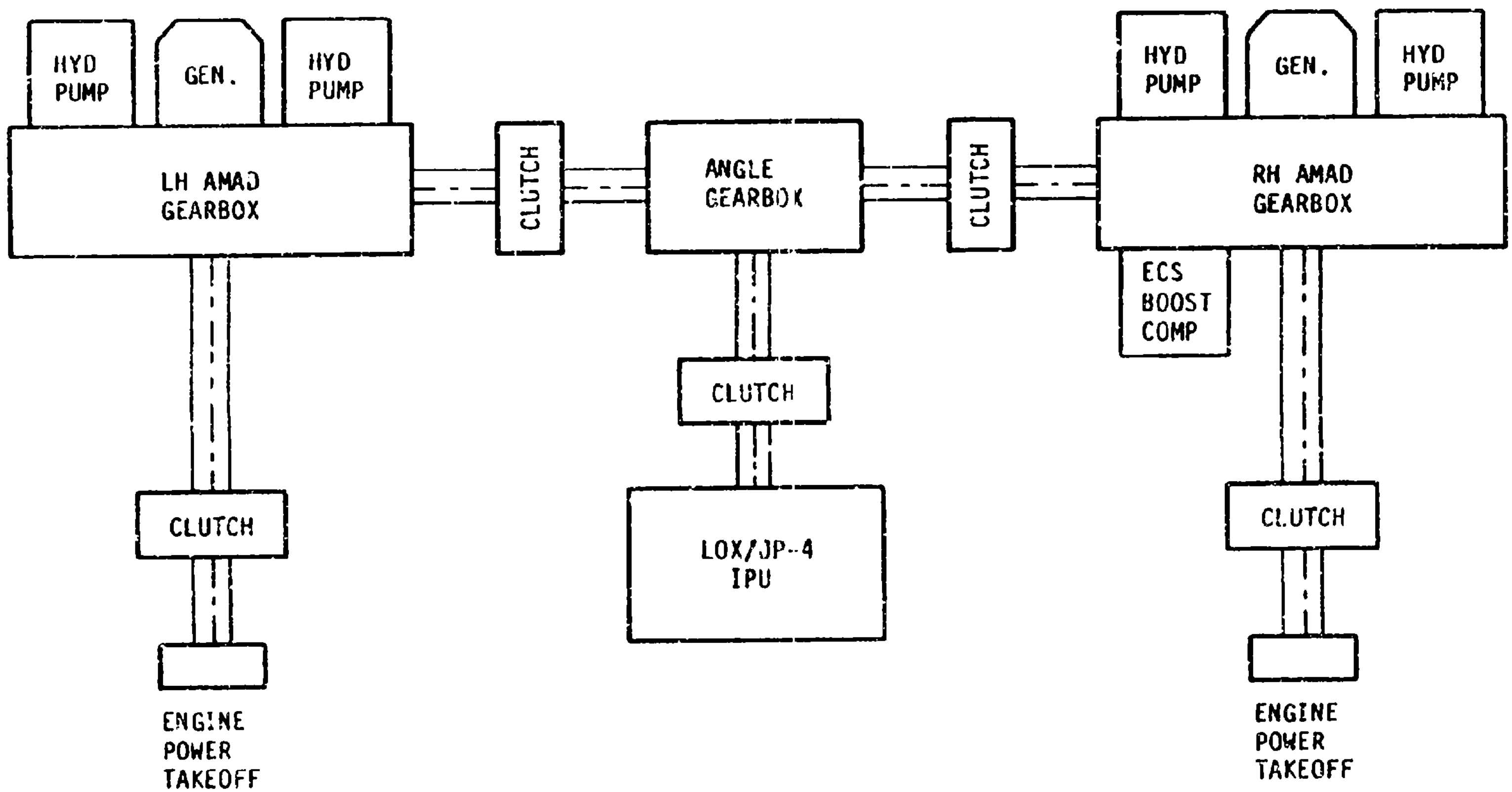


Figure 14 Secondary Power System Arrangement - Baseline Airplane

TABLE 9

ACCESSORY DRIVE SYSTEM COMPONENTS

ITEM		WEIGHT (POUNDS)
RH AMAD GEARBOX		60
RH INPUT CLUTCH		12
RH OUTPUT CLUTCH		7
RH INPUT SHAFTING		8
RH OUTPUT SHAFTING		3
RH STRUCTURAL PROVISIONS		9
	TOTAL, RH AMAD SYSTEM	99
LH AMAD GEARBOX		54
LH INPUT CLUTCH		12
LH OUTPUT CLUTCH		7
LH INPUT SHAFTING		8
LH OUTPUT SHAFTING		3
LH STRUCTURAL PROVISIONS		9
	TOTAL, LH AMAD SYSTEM	93
ANGLE AMAD GEARBOX		25
ANGLE BOX INPUT CLUTCH		7
ANGLE BOX INPUT SHAFTING		3
ANGLE BOX STRUCTURAL PROVISIONS		4
	TOTAL, ANGLE AMAD SYSTEM	39

(1) Integrated Drive Generator (IDG) system
(2) Cycloconverter type variable-speed, constant-frequency (VSCF) System
(3) DC-Link type VSCF system

The cycloconverter type VSCF concept was selected because of its higher operating efficiency, lower life-cycle cost, and higher reliability. Equipment rating is based on the electrical load analysis discussed in the following paragraph.

3.10.2.1 Load Analysis

A detailed electrical load analysis was conducted during Phase II and is shown in Figure 15 and Tables 10 and 11.

3.10.2.2 Selected System Arrangement

A schematic diagram of the electrical power system arrangement is shown in Figure 16 and a list of major components and weights in Table 12. Primary power generation consists of two samarium-cobalt permanent-magnet generators, one mounted on each AMAD gearbox, as shown in Figure 14. Permanent-magnet generators were selected rather than wound rotor generators because of increased generator efficiency, improved reliability, no rotor cooling requirement, and improved rotor balance due to the solid rotor. The variable-frequency generator output is fed to a cycloconverter, the output of which is 3-phase 120/208 volts, 400 Hz. Each generator/cycloconverter channel is rated at 60 kVA to provide margin for load growth. The AC load buses are interconnected by switches which allow transferring loads of a disabled generator to the other generator. Logic prevents parallel operation of the generators. Three transformer-rectifier units (TRU) convert 3-phase 400 Hz power to 28 volts DC.

AC and DC ground buses permit ground servicing of the airplane and checkout of some equipment using ground power without energizing all of the equipment, particularly electronics, for long periods of time on the ground. The source of ground power can be either external electrical power via an external power receptacle or one of the AMAD gearbox-mounted main aircraft generators driven by the IPU.

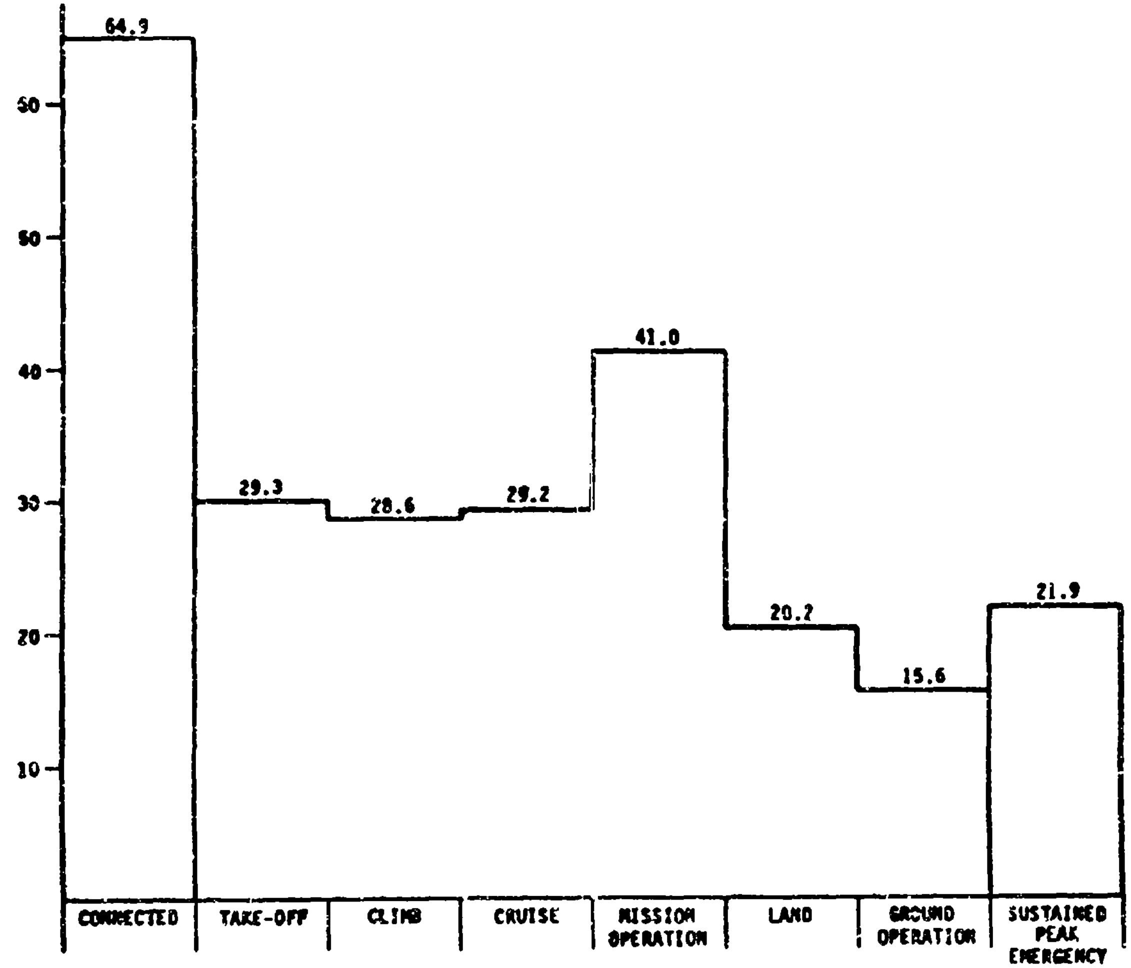


Figure 15 Electrical Load Profile - Baseline Airplane

TABLE 10 BASELINE AIRPLANE ELECTRICAL LOAD SUMMARY

SHEET 1 OF 1

ELECTRICAL LOAD TOTALS — ITEM DESCRIPTION	NO	BUS	CODE	MAX. CONNECTED LOADS, KW	UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, KW: TAKE OFF	CLIMB	CRUISE	MISSION OPERAT.	LAND	KW GROUND OPERATION	KW SUSTAINED PEAK EMERGENCY	REMARKS, OPERATING TIMES
TOTAL 115 VAC POWER				56.73	25.70	25.20	26.08	37.08	17.10	12.81	18.53	
(TABLE 11 SHEET 5)												
TOTAL 28VDC POWER				8.20	3.57	3.44	3.16	3.94	3.11	2.76	3.37	
(28 V INPUT)												
(TABLE 11 SHEET 5)												
TOTAL 115VAC + 28VDC				64.93	29.27	28.64	29.24	41.02	20.21	15.57	21.90	
TOTALS												

TABLE 11 BASELINE AIRPLANE ELECTRICAL LOAD ANALYSIS

SHEET 1 OF 5

AIR VEHICLE AND AVIONICS LOADS			CODE	MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS, GROUND OPERATION		WATTS, SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT.		LAND						
ITEM DESCRIPTION	NO	BUS		115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	
RELAYS, CONTACTORS, CONTROLS			NO	300	750	250	500	250	500	200	450	200	600	200	500	150	300	150	400	
INSTRUMENTATION			[illegible]	100	150	100	150	100	150	100	150	300	150	100	150	100	150	100	150	
PRIMARY FUEL BOOST PUMPS	2		NO	7300		7300		7000		6000		6500		5000		5000		5000		
BACK-UP FUEL BOOST PUMPS	2		NO	7300		6000														
DEFUELING CONTROL	[illegible]		NO		150															MOMENTARY MINOR LOAD
FUEL TRANSFER PUMPS			NO	7300				[illegible]		5000		5000								
FUEL TRANSFER CONTROLS			NO		200		50		50		50		50		50		50		50	
FUEL DUMP SYSTEM			NO		150															MOMENTARY MINOR LOAD
FUEL TANK INERTING CONTROL			NO		100								70							
ENGINE START CONTROLS	2		NO		150												150			
FLY-BY-WIRE COMPUTER	4		YES		260		260		260		260		260		260		260		260	
DATA PROCESSORS	6		YES																	
MEMORY	3		YES																	
DISPLAYS & CONTROLS	1		YES																	
FLIGHT CONTROL (FBW) COMPUTERS	3		YES		750		750		750		750		750		750		750		750	
INTEGRATED INFORMATION MANAGEMENT SYSTEM - IIMS				5400		5400		5400		5400		5400		5400		5400		5400		
NAVIGATION LIGHTS	4		NO	270		220		220		220		220		220				220		
COCKPIT & INSTRUMENTATION LIGHTS			NO	250		200		200		200		200		200		200		150		
LANDING LIGHTS	2		NO	600		600								600				600		
[illegible] Factors Have [illegible] Applied	SUB-TOTALS			28,770	2660	20,070	1710	18,170	1710	17,[illegible]20	1660	17,620	1 30	11,720	1710	10,850	1610	11,620	1610	

TABLE 11 BASELINE AIRPLANE ELECTRICAL LOAD ANALYSIS

AIR VEHICLE AND AVIONICS LOADS — ITEM DESCRIPTION	NO	BUS	CODL	MAX. CONNECTED LOADS, WATTS — 115V 400 HZ.	MAX. CONNECTED LOADS, WATTS — 28V DC	TAKE OFF — 115V 400 HZ.	TAKE OFF — 28V DC	CLIMB — 115V 400 HZ.	CLIMB — 28V DC	CRUISE — 115V 400 HZ.	CRUISE — 28V DC	MISSION OPERAT. — 115V 400 HZ.	MISSION OPERAT. — 28V DC	LAND — 115V 400 HZ.	LAND — 28V DC	WATTS, GROUND OPERATION — 115V 400 HZ.	WATTS, GROUND OPERATION — 28V DC	WATTS, SUSTAINED PEAK EMERGENCY — 115V 400 HZ.	WATTS, SUSTAINED PEAK EMERGENCY — 28V DC	REMARKS, OPERATING TIMES
AIR DATA SYSTEM HEATERS	1		NO	1500		1000		1000						1000				1200		
TOTAL TEMPERATURE PROBE HEATERS	1		NO	270		270		270						270						
GYRO SYSTEMS HEATERS	2		NO	100		50		50		50		50		50		50				
BATTERY HEATER	1		NO	300		100				50		50		50		100				
WINDSHIELD HEATERS	1		NO	2500		2000		2000						2000				2000		
WINDSHIELD WIPERS					200		200								200				200	
ECS SYSTEMS CONTROLS	2		NO		150		30		30		30		30		30		30		30	
CABIN PRESSURE CONTROLS			NO		100		20		20		20		20		20		20		20	
ELECTRONIC COOLING LIQUID PUMPS	2		NO	2000		1000		1000		1000		1000		1000		1000		1000		
BATTERY CHARGING			NO		900		450		250		50		50				50		50	
OXYGEN SYSTEMS CONTROLS			NO		15						15		15						15	
HYDRAULIC SYSTEMS CONTROLS			NO		30		30		30		30		30		30		30		30	
FIRE DE[illegible]TION, WARNING & EXTINGUISHING	[illegible]		[illegible]		25		25		25		25		25		25		25		25	
LEADING EDGE FLAPS ACTUATOR CONTROLS	2		NO		15		5		5						5				5	
NOSE WHEEL STEERING CONTROLS	1		NO		15		10								10				10	
LANDING GEAR CONTROLS	3		NO		30				5						5				5	
ANTI-SKID SYSTEM	1		NO		15										10				10	
[illegible] factors have [illegible] applied — SUB-TOTALS				6670	[illegible]95	4420	770	4320	365	1100	170	1200	170	4370	385	1150	145	4200	400	

TABLE 11 BASELINE AIRPLANE ELECTRICAL LOAD ANALYSIS

SHEET 3 OF 5

[illegible] COMMUNICATIONS, NAVIGATION & IDENTIFICATION LOADS — ITEM DESCRIPTION	NO.	ZUS	C U O L	CONNECTED LOADS, WATTS 115V 400 HZ.	CONNECTED LOADS, WATTS 28V DC	TAKE OFF 115V 400 HZ.	TAKE OFF 28V DC	CLIMB 115V 400 HZ.	CLIMB 28V DC	CRUISE 115V 400 HZ.	CRUISE 28V DC	MISSION OPERAT. 115V 400 HZ.	MISSION OPERAT. 28V DC	LAND 115V 400 HZ.	LAND 28V DC	WATTS, GROUND OPERATION 115V 400 HZ.	WATTS, GROUND OPERATION 28V DC	WATTS, SUSTAINED PEAK EMERGENCY 115V 400 HZ.	WATTS, SUSTAINED PEAK EMERGENCY 28V DC	REMARKS, OPERATING TIMES
JTIDS/TACAN/SIF	1		YES	700		700		700		700		700		700		700		700		
UHF/VHF RADIO	1		YES		50		50		50		50		50		50		50		50	
PARABOLIC FILTER CORNETT CLEAR	1		YES		160		160		160		160		160		160		160		160	
VOICE PROCESSOR	1		NO	5		5		5		5		5		5		5		5		
KIT-1 A/T	1		YES	37		37		37		37		37		37		37		37		
RECORDERS	1		YES		240		240		240		240		240		240		240		240	
WIDE BAND RECEIVER	1		NO		80		80		80		80		80		80		80		80	
GPS (GLOBAL POSITIONING SYS)	1		YES	200		200		200		200		200						200		
MIRS (MULTI-FUNCTION INERTIAL REFERENCE)	1		YES	200		200		200		200		200		200				200		
PLS/ILS (LANDING SYSTEM)	1		NO		20		20		20		20		20		20				20	
AIR DATA COMPUTER	1		NO	65		65		65		65		65		65		65		65		
[illegible]	TOTALS			1207	550	1207	550	1207	550	1207	550	1207	550	1007	550	807	530	1207	550	

TABLE 11 BASELINE AIRPLANE ELECTRICAL LOAD ANALYSIS

SHEET 4 OF 5

ARMAMENT/PAYLOAD LOADS			COOL	MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS, GROUND OPERATION		WATTS, SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT		LAND						
ITEM DESCRIPTION	NO	JUS		115V 400 HZ	28V DC	115V 400 HZ	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	
GUN CONTROLS	1		NO	3606	812							TRANSIENT .05 SEC. SURGES								.05 SEC. SURGES
MISSILE LAUNCHER	1		NO		448							TRANSIENT .05 SEC. SURGES								.06 SEC. SURGES
XPANDABLES DISPENSER	2		NO		206								50							
SYNTHETIC APERTURE SAR RADAR	1		YES	5000						5000		5000								
IRCM	1		YES	2000								2000								
PASSIVE SENSORS (IR & EO)	1		YES	1500	300			1500	300	1500	300	1500	300					1500	300	
WARNING PROCESSOR	1		YES	INCLUDED IN ABOVE LOAD DATA																
ECM TRANSMITTERS	2		YES	6000	300							6000	300							
ECM XMTR CONTROL	1		NO		150								150							
FIRE CONTROL AND STORES MANAGEMENT	1		NO	150						150		150								
WEAPONS ARMING	1		NO	50	50							TRANSIENT 2.0 SEC. SURGES								2.0 SEC. SURGES
WEAPONS HEATERS	1		NO	1000								1000								
TARGET ACQUISITION INFRARED RADAR			YES	1500								1500								
[illegible] TOTALS				20,800	2260			1500	300	6650	300	17,150	800					1500	300	

TABLE 11 BASELINE AIRPLANE ELECTRICAL LOAD ANALYSIS

SHEET 5 OF 5

SUMMARY & TOTAL POWER				MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS, GROUND OPERATION		WATTS, SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT.		LAND						
ITEM DESCRIPTION	NO	BUS	COOL	115V 400 HZ	28V DC	115V 400 HZ	28V DC	115V 400 HZ	28V DC	115V 400 HZ	28V DC	115V 400 HZ	28V DC	115V 400 HZ	28V DC	115V 400 HZ	28V DC	115V 400 HZ	28V DC	
AIR VEHICLE AVIONICS SHEET 1				26,770	2,660	20,070	1,710	18,170	1,710	17,120	1,555	17,620	1,830	11,720	1,710	10,860	1,463	11,620	1,610	
AIR VEHICLE AVIONICS SHEET 2				6,670	1,495	4,420	770	4,370	365	1,100	170	1,100	170	4,370	385	1,150	355	4,200	400	
AIR VEHICLE AVIONICS TOTAL				36,440	4,155	24,490	2,480	22,490	2,073	18,220	1,830	18,720	2,000	16,090	2,095	12,000	1,815	15,820	2,016	
COMMUNICATIONS, NAVIGATION, & IDENTIFICATION (CNI)				1,207	550	1,207	550	1,207	550	1,207	550	1,207	550	1,007	550	807	530	1,207	550	
ARMAMENT & PAYLOAD				20,000	2,260			1,500	300	6,650	700	[illegible]	800					1,500	300	
TOTAL AC POWER				54,727		25,6[illegible]		25,197		26,077		37,077		17,097		12,807		18,527		
TOTAL DC POWER					6,965		3030		2925		2640		3350		2645		2,345		2060	
TRU LOSSES					1230		535		515		475		590		465		415		506	
TOTAL TRU INPUT				8195		3565		3440		3155		3940		3110		2,767		3365		
TOTAL AC & DC POWER				64,922		29,262		28,637		29,232		41,017		20,207		15,567		21,592		
* ...ity factors have been applied			TOTALS																	

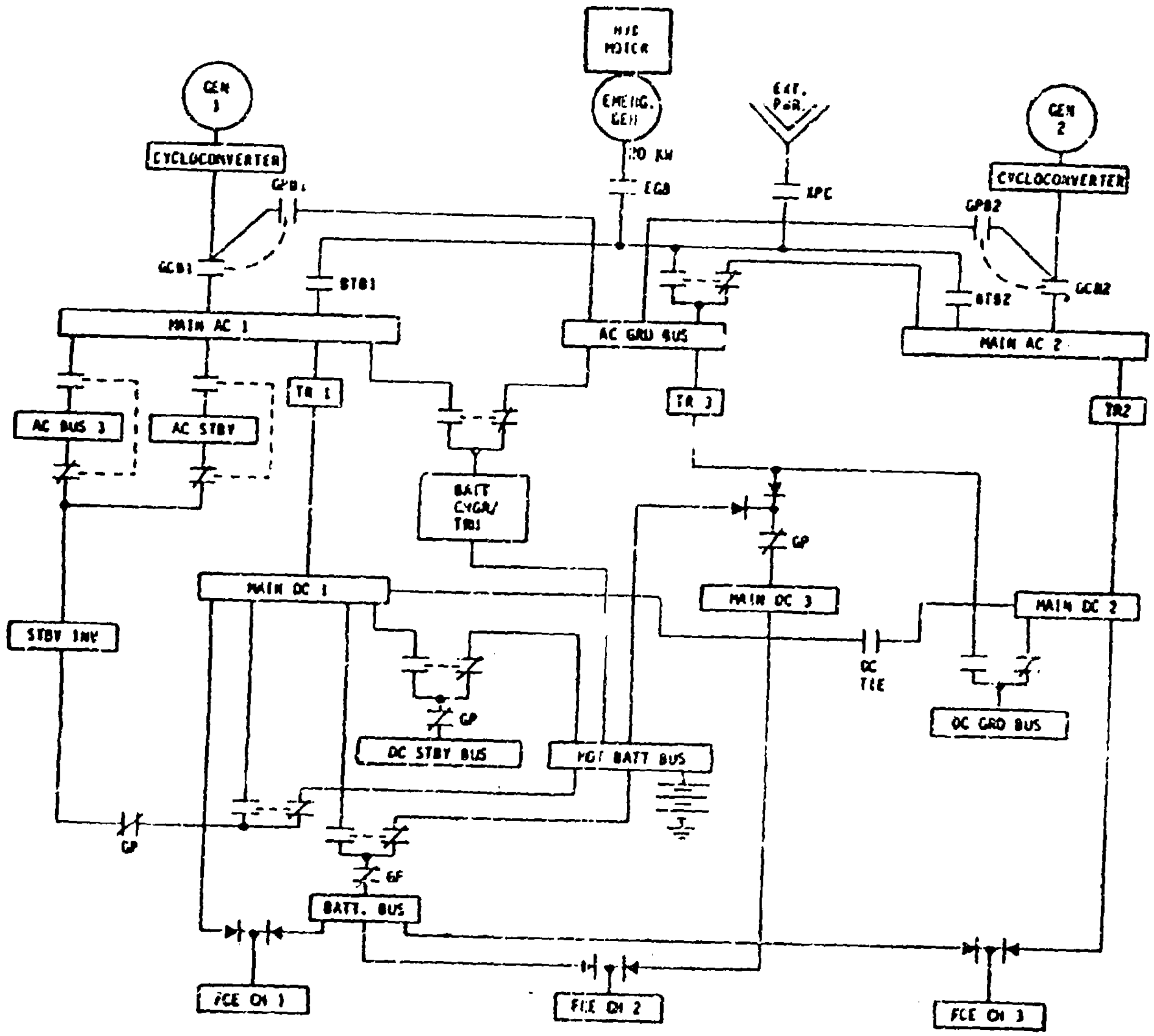


Figure 16 Electrical System Schematic - Baseline Airplane

TABLE 12

BASELINE AIRPLANE - ELECTRICAL POWER SYSTEM COMPONENTS

COMPONENT	QUANTITY	UNIT WEIGHT (lbs)	TOTAL WEIGHT (lbs)
Generator	2	30	60
Cycloconverter	2	60	120
Emergency Generator	1	36	36
Hyd. Motor-Emerg Gen	1	14.8	14.8
Control Valve-Emerg Gen	1	1.1	1.1
Transformer-Rectifier Unit	3	12.5	37.5
Battery 40 A-Hr	1	75	75
Battery Charger	1	6.8	6.8
Static Inverter	1	13.0	13.0
AC Power Relay 3 PDT	1	1.2	1.2
AC Power Relay 3 PDT	1	1.6	1.6
AC Power Contactor 3 PST, 20 kVA	1	3.8	3.8
AC Power Contactor 3 PST, 60 kVA	3	5.3	15.9
AC Power Contactor 3 PDT, 60 kVA	2	6.2	12.4
DC Power Contactor SPST	3	0.8	2.4
DC Power Contactor SPDT	2	2.1	4.2
Wiring and Connectors, total		123	123
		TOTAL	528.7

Three high-reliability DC buses are provided for powering the triple-redundant fly-by-wire flight control system. Each of these buses (FCE CH1, FCE CH2, and FCE CH3 in Figure 16) is supplied by two sources of power: a generator and a battery. The three buses share a common battery, but each bus is connected to the primary electrical power sources, i.e. the generators, through a different TRU. Since there are only two main generators, two of the TRUs have to share a common generator. One of these TRUs (number 3 in Figure 16) is supplied from the AC ground bus, which is provided with switching and control logic so that if either main AC bus is energized the AC ground bus is energized. Thus no single failure of a power source will cause a power interruption on any of the FCE buses. Loss of the battery and one generator will cause momentary loss of one or two FCE channels, depending on whether or not the failed generator is the one normally supplying the AC ground bus. Power will be recovered to all FCE buses within a few milliseconds when the AC bus or buses on the failed generator are transferred automatically to the remaining generator.

An emergency generator, driven at 8000 rpm by a 0.375-cipr hydraulic motor, is included to provide power for the critical electrical equipment such as the fly-by-wire flight controls in the event of loss of both main generators. This generator is rated at 20 kVA, 3-phase 120/208 volts 400 Hz. It can be connected to any or all of the three main AC buses.

A 40-ampere-hour nickel-cadmium battery is included as backup for the emergency generator. The battery serves to maintain continuity of power to the critical loads during start-up of the emergency generator or the IPU following loss of both main generators or both engines. In the event of loss of both engines, the IPU will be clutched to one or both AMAD gearboxes to drive the hydraulic pumps and the main generators. The IPU is capable of starting an engine in flight while driving the loaded generators and hydraulic pumps.

3.10.3 Hydraulic Power System

The primary goal in configuring the hydraulic power system for the Baseline Airplane was to provide the most competitive arrangement, in terms of size, weight, reliability, maintainability, and cost, that could be considered available for the 1990 time frame. One of the first questions was to determine the number of hydraulic subsystems required.

Rigorous compliance with MIL-H-5440G could lead to the use of three subsystems since it requires that the hydraulic system(s) be configured such that any two fluid system failures due to combat or other damage which cause loss of fluid or pressure will not result in complete loss of flight control, and that the surviving system(s) shall provide sufficient control to meet the level 3 flying qualities of MIL-F-8785 for conventional takeoff and landing. However, from the requirements for the individual actuation systems listed in Table 4, only the canard and elevon actuation systems have a firm requirement to maintain actuation capability after the failure of two power sources. Therefore, it was possible to consider either of two basic options:

a. Provide three main hydraulic subsystems
b. Provide two main subsystems with one or more additional auxiliary systems

Before a selection was made, a load analysis was conducted, operating pressure selected, and a number of configuration arrangements were made for study.

3.10.3.1 Load Analysis

The hydraulic flow rates required for each actuator and hydraulic motor to obtain its design slew rate or speed were determined during Phase II and are listed in Table 13. The maximum simultaneous flow demands for various flight conditions were determined for each of the candidate hydraulic systems and are listed in Tables 14 through 16 for the selected arrangement. Pump sizes were then determined and are listed in Table 17.

3.10.3.2 Operating Pressure

A number of studies, starting with those conducted by the Glenn L. Martin Company (published in 1954 in Reference 7) have shown that hydraulic system weight can be reduced by increasing system operating pressure above the standard 3,000 psi level. Several aircraft in the intervening years,

TABLE 13 ACTUATION LOADS AND HYDRAULIC FLOW REQUIREMENTS

Actuation Function	Max. Load T lb-ft F lb P hp	Max. Rate deg/sec v in/sec N rpm	Actuator No. & Type	Actuator Displacement P in^3/in V in^3/deg M in^3/rev	Actuator Max. Rate in/sec deg/sec rev/min	Hydraulic Flow Rate Per Actuator (Normal) gpm avg.	Hydraulic Flow Rate Per Actuator (Adaptive) gpm avg.
(2) Canards	±43,250T Total	70	(6) Piston	3.6 P	10.86	10.2	5.1
(2) Elevons	+19,200 -42,000 T/side	70	(8) Piston	4.25 P 8.5 P	4.8	8.0	4.0
(1) Rudder	±17,618T Total +3,534	75	(2) Motor	0.31 P 2.7 P	11,000	14.9	7.5
(4) Spoilers	-7,550 T/side	100	(4) Piston	3.0 P	6.0	4.5	
(6) L.E. Flaps	-101,400T Total	15	(12) Piston	1.0 P 7.4 P	0.83	0.9	
(2) Inlet Centerbodies	20,800F ea.	4V	(2) Piston	5.6 P 1.0 P	4.0	3.4	
(8) Bypass Doors	25T ea.	90	(4) Vane	0.025 V	90	0.6	
(2) Main Gear	15,000F/act.	6 sec.	(2) Piston	6.76 P 4.69 P	2.83	5.0 Gear Retract 3.5 Gear Extend	
(1) Nose Gear	13,330F/act.	6 sec.	(1) Piston	4.36 P 6.76 P	2.43	2.8 Gear Retract 4.3 Gear Extend	
(1) Gun Drive	25 hp	1,800 N	(1) Motor	0.34 M	7,200	11.2	
(1) ECS pack Compr.	10.7 hp	10,000 N	(1) Motor	0.10 M	10,000	4.5	
(1) Elex	42.9 hp	12,000 N	(1) Motor		8,000	18.2 Supersonic	
Co[illegible]an	21.5	6,000 N		0.52 M	4,000	9.1 Subsonic	

Note: N.G. Steering, Wheel Brakes, Canopy, Aerial Refueling, and Emergency Generator Drive loads not shown. They do not enter into pump sizing.

TABLE 14 ACTUATION RATE REQUIREMENTS AND HYDRAULIC FLOW DEMANDS

TOTAL AIRCRAFT REQUIREMENTS

Actuation Function	Number of Actuators or Valves	Max. Flow per Cycle	Takeoff and Climb	Subsonic Cruise	Supersonic Cruise	Weapon Delivery	Subsonic Strafing	Descent and Land
		gpm	% activity and gpm	%/gpm	%/gpm	%/gpm	%/gpm	%/gpm
Canards	6	30.6	100/30.6	75/23	75/23	75/23	75/23	70/21.4
Elevons	8	32.0	70/22.4	75/24	25/8	75/24	75/24	100/32
Rudder	2	7.5	100/7.5	50/3.7	50/3.7	50/3.7	50/3.7	50/3.7
Spoilers	4	18.0	50/9.0	25/4.5	—	—	50/9.0	50/9.0
LE Flaps	12	10.8	20/2.2	50/5.4	—	—	100/10.8	50/5.4
Inlet	2	6.8	—	—	100/6.8	100/6.8	—	—
Centerbody								
Bypass	4	2.4	—	—	100/2.4	100/2.4	—	—
Doors								
Main Gear	2	Ret 10.0	10.0					7.0
		Ext 7.0						
Nose Gear	1	Ret 2.8	2.8					
		Ext. 4.3						4.3
Gun Drive	1	11.2					11.2	
ECS	1	4.5	4.5	4.5	4.5	4.5	4.5	4.5
Compr.								
ECS Fan	1	Subs 9.1	9.1	9.1			9.1	9.1
		Sup 18.2			18.2	18.2		
Valve	30/36	7.2	6.0	6.0	7.2	7.2	6.0	6.0
Leakage								
Total		162.0	104.1	80.2	73.8	89.8	101.3	102.4

TABLE 15 ACTUATION RATE REQUIREMENTS AND HYDRAULIC FLOW DEMANDS

SUBSYSTEM 1 OF 3 SUBSYSTEMS

Actuation Function	Number of Actuators or Valves	Max. Flow per Cycle	Takeoff and Climb	Subsonic Cruise	Supersonic Cruise	Weapon Delivery	Subsonic Strafing	Descent and Land
		gpm	% activity and gpm	%/gpm	%/gpm	%/gpm	%/gpm	%/gpm
Canards	2	10.2	100/10.2	75/7.6	75/7.6	75/7.6	75/7.6	70/7.1
Elevons	4	16	70/11.2	75/12	25/4	75/12	75/12	100/16
Rudder								
Spoilers								
LE Flaps								
Inlet								
Centerbody								
Bypass								
Doors								
Main Gear	2	Ret 10.0	10.0					7.0
		Ext 7.0						
Nose Gear	1	Ret 2.8	2.8					
		Ext. 4.3						4.3
Gun Drive	1	11.2					11.2	
ECS	1	4.5	4.5	4.5	4.5	4.5	4.5	4.5
Compr.								
ECS Fan	1	Subs 9.1	9.1	9.1			9.1	9.1
		Sup 18.2			18.2	18.2		
Valve	6	1.2	1.2	1.2	1.2	1.2	1.2	1.2
Leakage								
Total			49.0	34.4	35.5	43.5	45.6	49.2

TABLE 16 ACTUATION RATE REQUIREMENTS AND HYDRAULIC FLOW DEMANDS

SUBSYSTEMS 2 AND 3 OF 3 SUBSYSTEMS

Actuation Function	Number of Actuators or Valves	Max. Flow per Cycle	Takeoff and Climb	Subsonic Cruise	Supersonic Cruise	Weapon Delivery	Subsonic Strafing	Descent and Land
		gpm	% activity and gpm	%/gpm	%/gpm	%/gpm	%/gpm	%/gpm
Canards	2	10.2	100/10.2	75/7.6	75/7.6	75/7.6	75/7.6	70/7.1
Elevons	2	6.0	70/5.6	75/6.0	25/2.0	75/6.0	75/6.0	100/8.0
Rudder	1	3.7	100/3.7	50/1.8	50/1.8	50/1.8	50/1.8	50/1.8
Spoilers	2	9.0	50/4.5	25/2.2	—	—	50/4.5	50/4.5
LE Flaps	6	5.4	20/1.1	50/2.7	—	—	100/5.4	50/2.7
Inlet	1	2.2			100/3.4	100/3.4		
Centerbody								
Bypass	2	1.2			100/1.2	100/1.2		
Doors								
Main Gear								
Nose Gear								
Gun Drive								
ECS Compr.								
ECS Fan								
Valve	12/15	3.0	2.4	2.4	3.0	3.0	2.4	2.4
Leakage								
Total			27.5	22.7	19.0	23.0	27.7	26.5

TABLE 17 REQUIRED HYDRAULIC PUMP SIZES

THREE-SUBSYSTEM ARRANGEMENT

SYSTEM 1	(2)	43-GPM ENGINE-DRIVEN PUMPS
SYSTEM 2	(1)	46-GPM ENGINE-DRIVEN PUMP
SYSTEM 3	(1)	46-GPM ENGINE-DRIVEN PUMP

including the USAF B-70 and B-1 bombers, the Concorde supersonic transport, and other foreign aircraft, have been designed with 4,000 psi systems; and, the Navy, in their desire for absolute weight minimization for future V/STOL aircraft, has sponsored development of 8,000 psi system technology.

However, in studies previously conducted at Boeing, it was concluded that, with normal design practice for Air Force combat aircraft, the minimum weight of hydraulic transmission line tube runs would be obtained with a system operating pressure of approximately 5,000 psi and that their weight would increase at higher pressures. This is shown in Figure 17. As shown in the LAMINAR (F=4) curve, the minimum-weight pressure for tubing designed for laminar flow, with a burst safety factor of four times working pressure, is approximately 5,000 psi. With a burst safety factor of three times working pressure (the LAMINAR F=3 curve) the minimum-weight pressure is approximately 6,000 psi; however, there is very little reduction of weight by going to pressures above 5,000 psi.

These curves also show that the minimum-weight pressure increases if the tubing is sized for turbulent flow. Since most Navy aircraft are not required to start up from a cold soak condition and become airborne within a few minutes, as required for most Air Force combat aircraft, the Navy's tubing sizes can be smaller and the fluid flow is nearly always turbulent. (Note that Figure 17 was prepared for a presentation to the Naval Air Development Center and the Naval Air Systems Command, and that the curves are based on equations which included the characteristics of MIL-H-83282 fluid and the 3Al-2.5V titanium alloy tubing. It is expected that the minimum-weight pressures would be approximately the same for other hydraulic fluids but would be somewhat lower for tubing alloys with lower strength-to-weight ratios. However, for an ATS aircraft in the 1990 time frame, the use of 3Al-2.5V cold worked titanium tubing is considered a good choice at this time.)

Figure 18 illustrates the transition temperatures where laminar flow of MIL-H-5606 fluid in system tubing changes to turbulent flow for a typical design flow velocity of 20 feet per second. Note that for almost all of the normally used tubing sizes (-12 and smaller), the transition temperature is above zero degrees Fahrenheit. Since it is considered that the ATS aircraft

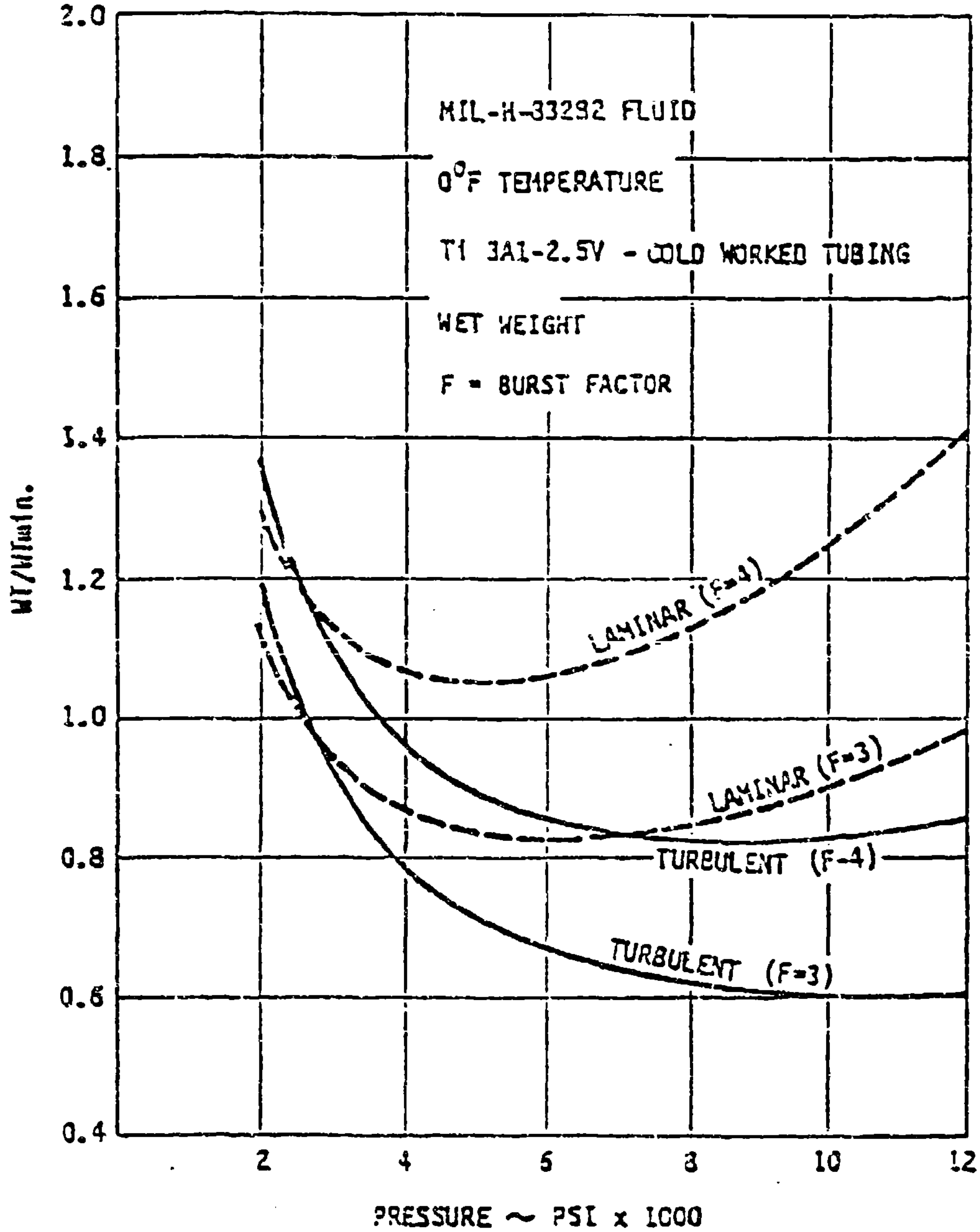


Figure 17 The Comparison of Relative Transmission Line Weight VS Hydraulic System Operating Pressure

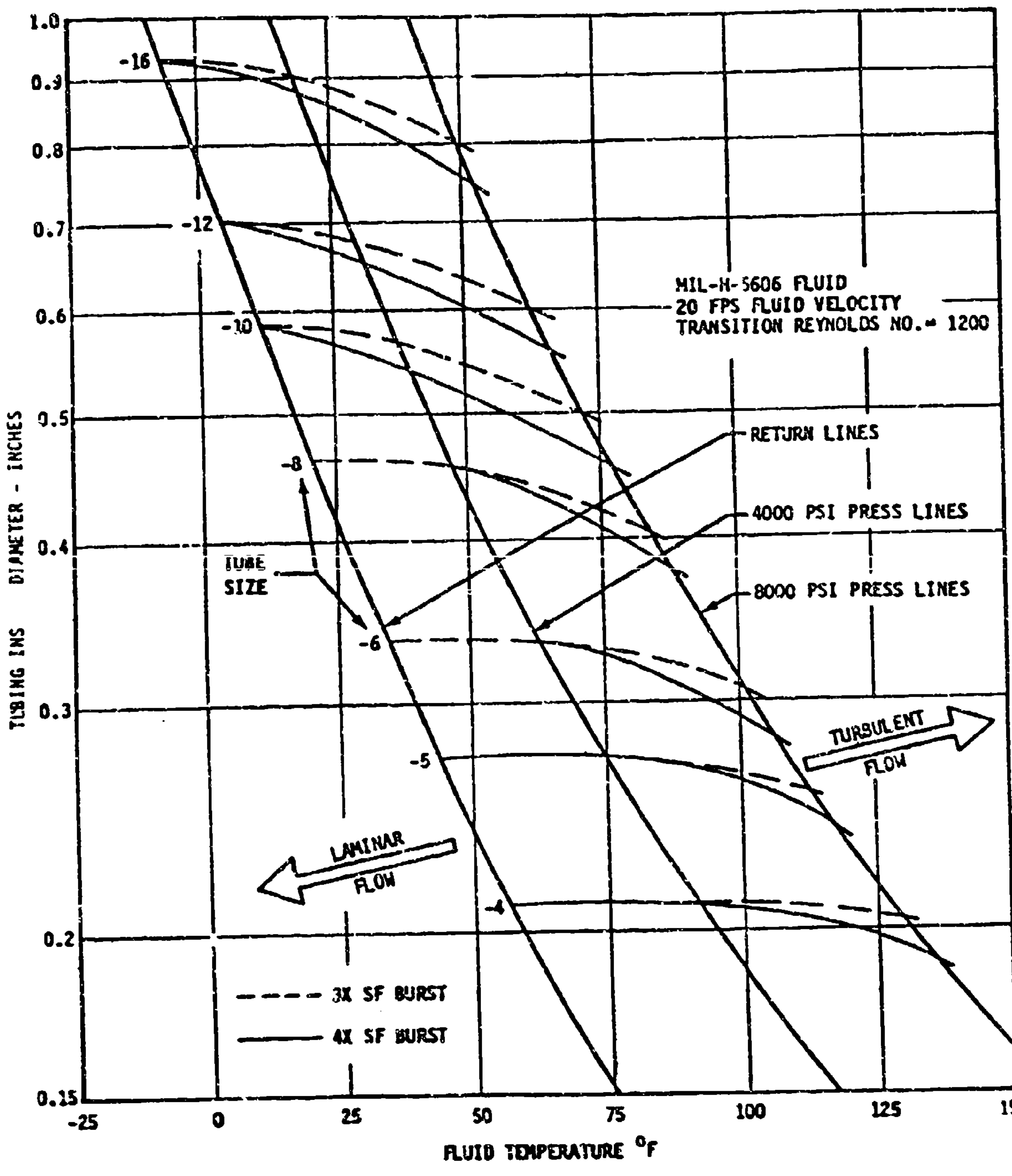


Figure 18 Laminar-Flow-to-Turbulent-Flow Transition Temperatures for Typical System Tube Sizes

used in this study is the type which must be able to start up from a cold soak condition and become airborne within minutes, it is assumed that there will be times when design flow rates must be provided at fluid temperatures below zero degrees and that the tubing must be designed for laminar flow conditions.

In addition to the transmission line tubing, the hydraulic actuators also represent a significant portion of the overall system weight. As shown in Figure 19, minimum weight for typical actuators is expected between 3,000 and 6,000 psi depending upon actuator force size. As shown in Figure 20, the optimum pressure for minimum space volume is somewhat higher, and also increases with actuator force size.

Therefore, in consideration that the predicted actuation forces for the study aircraft are high, and in the interests of weight and space optimization, 5,000 psi was chosen as the system operating pressure.

3.10.3.3 Selected System Arrangement

The three-system hydraulic power arrangement was selected for the following reasons:

(1) Hydraulic pump sizes required are within the range of sizes currently available for 3000 and 4000-psi aircraft hydraulic systems. The development of 5000-psi pumps in those sizes for use in the 1990-plus time frame should present no insurmountable problems for the pump manufacturers.
(2) The required sizes of the auxiliary pumps in the two system arrangements present a major problem due to the size of the electric drive motors.
(3) The three-system arrangement is lighter and less complex than the two-system arrangement.

A block diagram of the selected arrangement is shown in Figure 21, a schematic diagram in Figure 22, and a list of major components in Table 18.

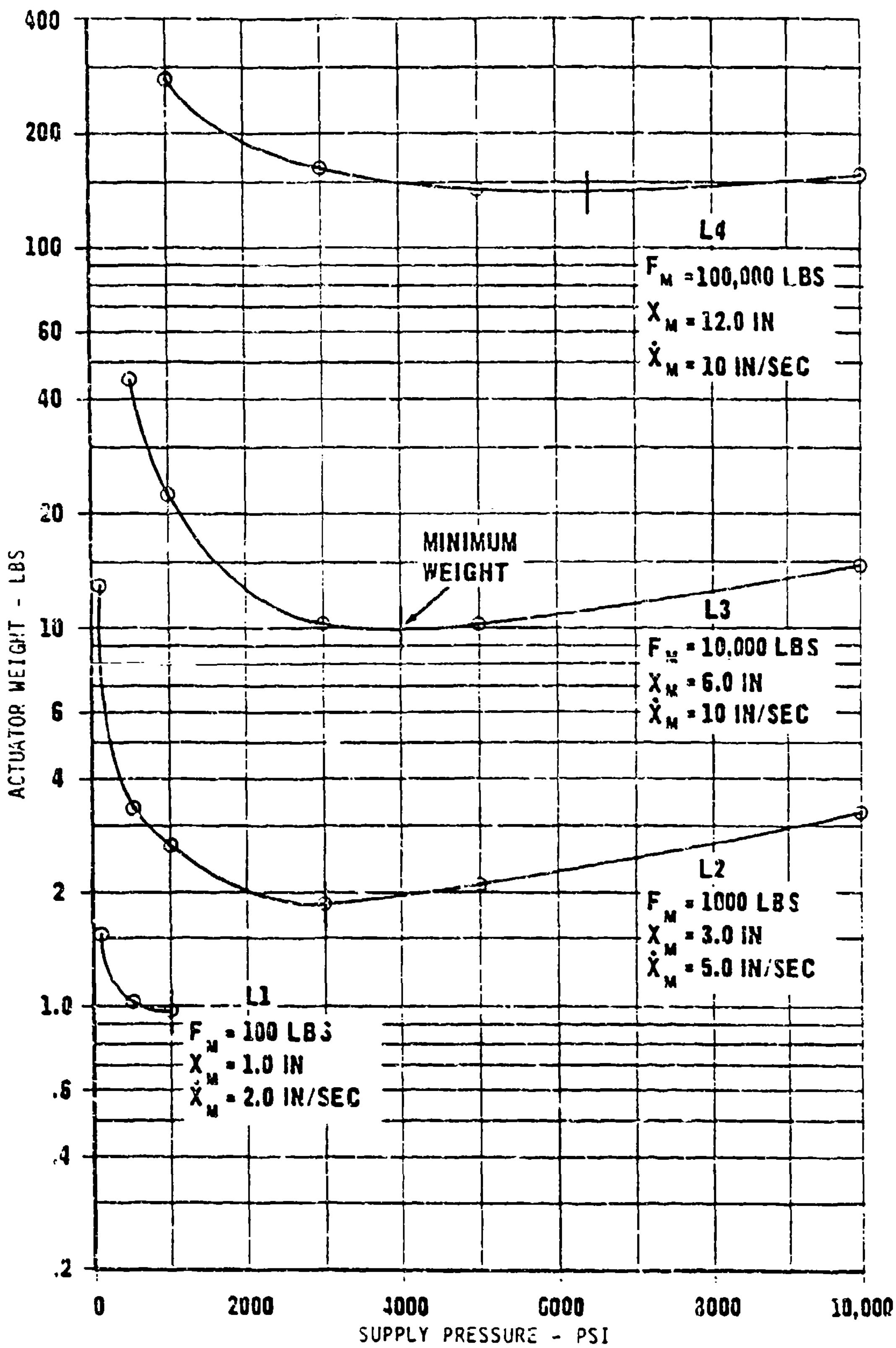


Figure 19 Comparative Actuator Weight
VS Hydraulic System Operating Pressure

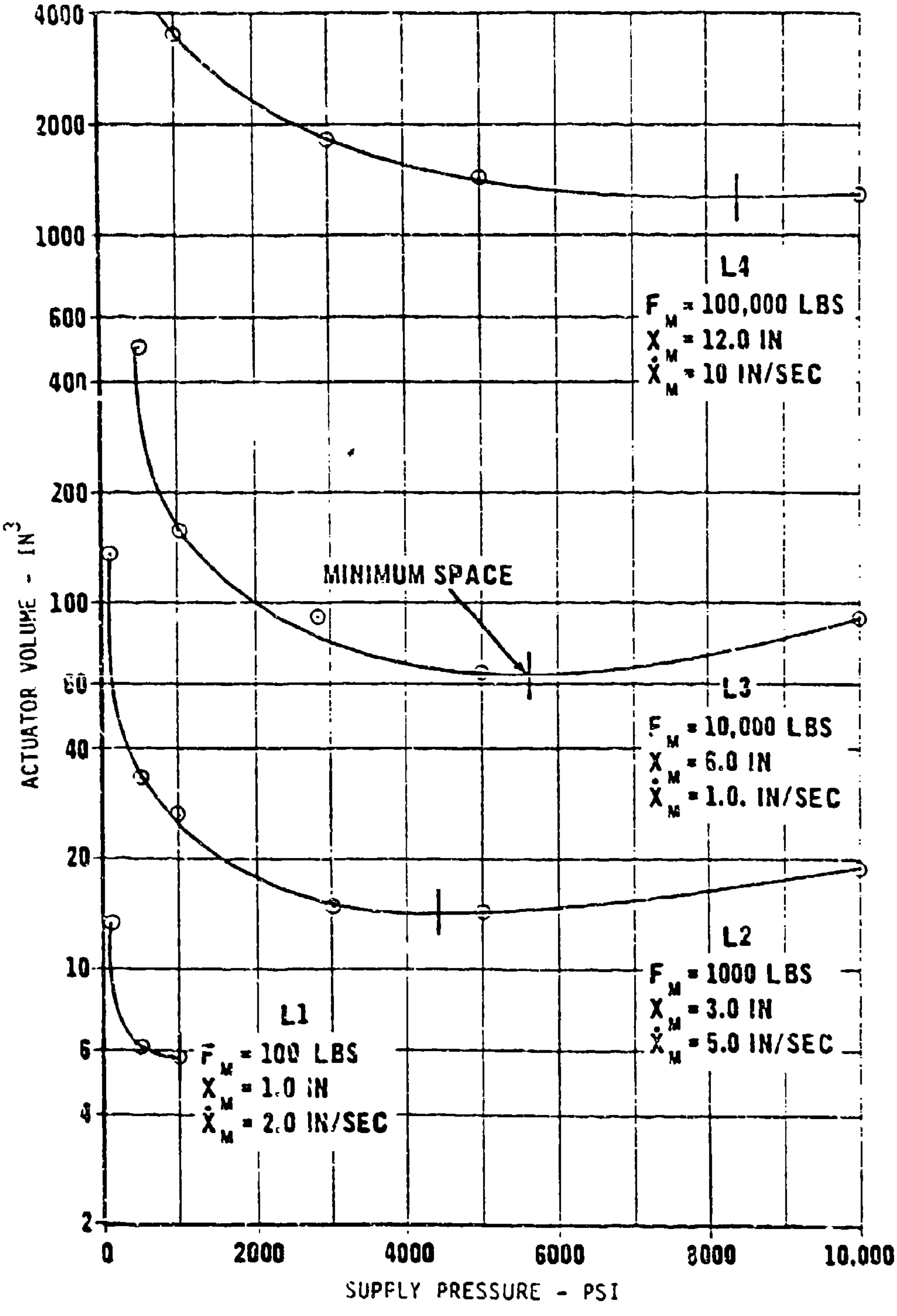


Figure 20 Comparative Actuator Volume
VS Hydraulic System Operating Pressure

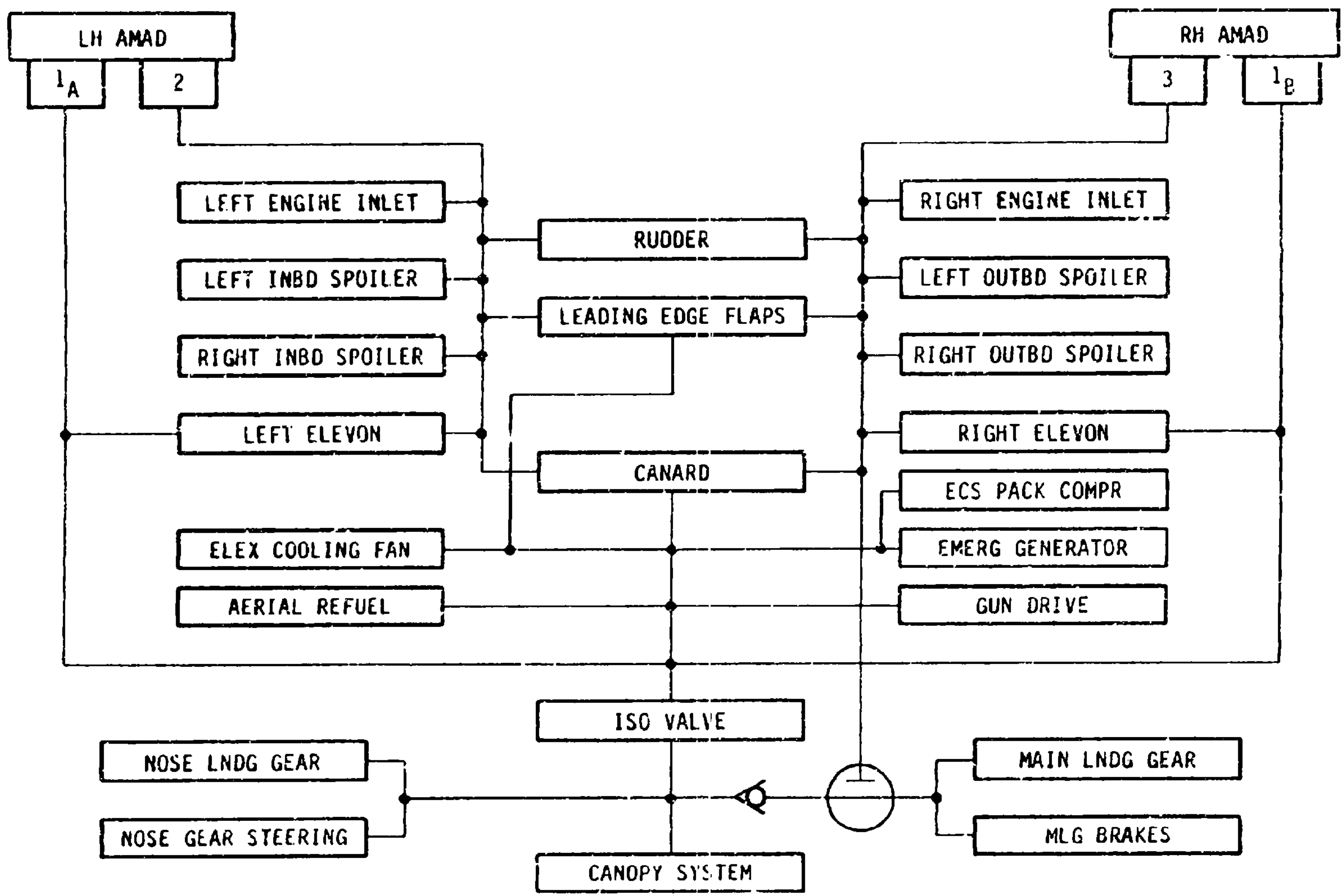


Figure 21 Hydraulic Power System Arrangement

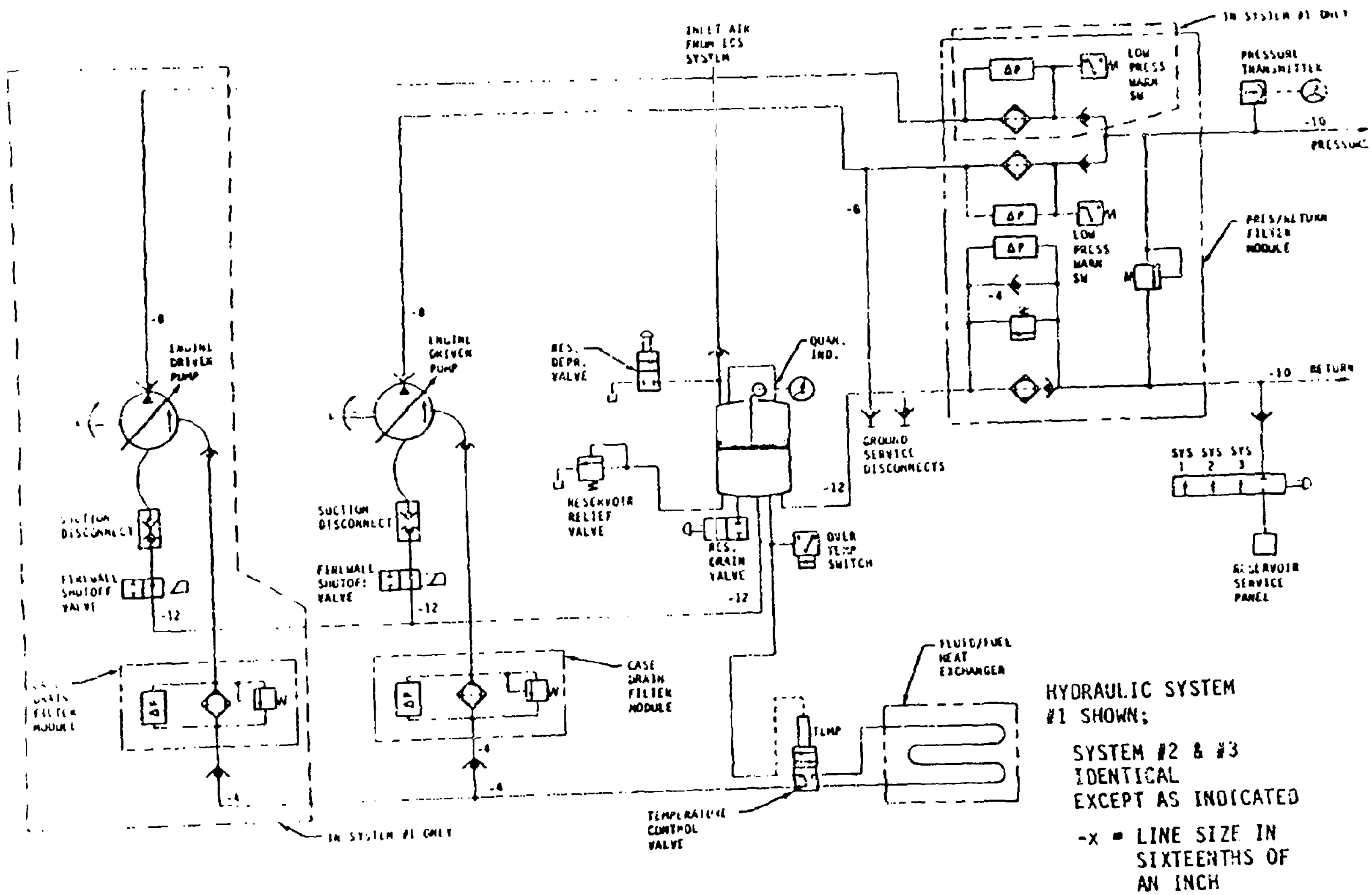


Figure 22 Hydraulic Power System Schematic

TABLE 18

BASELINE AIRPLANE - HYDRAULIC POWER SYSTEM COMPONENTS

COMPONENT	QUANTITY	UNIT WEIGHT (lbs)	FLUID WT PER UNIT	TOTAL WEIGHT (lbs)
Hydraulic Pump	4	27.0	3.0	120.0
Reservoir No 1	1	11.5	15.0	26.5
Reservoir No 2 and 3	2	5.0	6.0	22.0
Temp Control Valves	3	1.0	--	3.0
Over Temp Switches	3	0.1	--	0.3
Heat Exchangers	3	3.0	0.1	9.3
Filter Module No 1	1	23.0	2.3	25.3
Filter Module No 2 and 3	2	15.0	1.5	33.0
Case Drain Filter Module	4	8.0	0.4	33.6
Reservoir Service Panel	1	10.0	0.6	10.6
Reservoir Relief Valves	6	0.1	--	0.6
Reservoir Bleeder Valves	6	0.1	--	0.6
Firewall S.O. Valves	4	1.7	--	6.8
Disconnects	10	1.28	--	12.8
Hydraulic Hand Pump	1	3.4	--	3.4
Pressure Transmitters	3	0.2	--	0.6
Tubing and Fittings (Total)		80.8	52.3	133.1
			TOTAL	441.5

IV ALL-ELECTRIC AIRPLANE CONFIGURATION

4.1 General

The objective of the design phase was to select the most competitive combination of electrical actuation systems and electrical power systems for transmitting power to those systems and for providing fly-by-wire control to the flight control actuation systems that could be considered for the 1990-plus time frame. In keeping with the overall objectives and requirements, it was required that the selected electrical power system derive its power primarily from the engine through engine-driven electrical generators and transmit that power through a distribution system of electrical buses. The total secondary power system and actuation systems are defined so that a direct comparison can be made with the Baseline Airplane design described in Section III.

4.2 Actuation Systems for the All-Electric Airplane

Two actuation types were considered for the All-Electric Airplane actuation functions, i.e., the electromechanical actuator (EMA) system and the integrated actuator package (IAP) system. Three EMA schemes were considered: the servomotor gearbox, clutched electrical actuation, and the mechanical servo power package (MSPP). Also, three IAP concepts were considered: the servopump concept, accumulator stored-energy concept, and the fixed-displacement pump concept. The IAP concept, however, was rejected for all actuation functions since in each case it proved to be heavier than the comparable EMA in most applications.

Under a subcontract, AiResearch Manufacturing Company of California assisted in providing data for configurations of EMAs for the various actuation functions. The results of their study effort is reported in AiResearch Document No. 80-17284 (Reference 1).

Data obtained from AiResearch along with data obtained from other suppliers was used to arrive at a selection for the actuation system for each of the functions. Table 19 summarizes the selected systems for the airplane flight

TABLE 19 ALL-ELECTRIC AIRPLANE ACTUATION SUMMARY - CONTROL SURFACE

		Peak Motor HP		Controller/Inverter Cooling	
Actuator Function	Actuator Type	Total	Per Motor	Required	Method
Canard	Linear Ballscrew EMA	68	17	Yes	Cold Plate
Elevon	PDU & Hingeline Gearbox EMA	62	31	Yes	Cold Plate
Rudder	PDU & Hingeline Gearbox EMA	22	11	Yes	Cold Plate
Spoiler	Hingeline Motor and Gearbox EMA	18	9	Yes	Cold Plate
LE Flaps	Hingeline Motor and Gearbox EMA	68	11.4	Yes	Cold Plate
Engine Inlet Centerbody	Linear Ballscrew EMA	22	22	Yes	Cold Plate
Engine Inlet Bypass Doors	Motor-Gearbox EMA	0.12	0.06	Yes	Convection

controls and Tables 20 and 21 for the non-flight control functions. Figure 23 shows the location of the actuation systems in the aircraft and Figure 24 shows how these actuators are integrated into the aircraft. Each of the individual applications is covered in the following paragraphs.

For each actuator application that utilizes a DC brushless motor, a separate controller/inverter is required. During Phase II, various methods for packaging and cooling these units were investigated. The original packaging concept decided upon was an evaporative cooled configuration in which the electronics were installed in a circular container filled with a fluid cooling medium. However, after sizing the various controller/inverters to the individual actuation requirements, it was found that the units were very heavy, with approximately half the weight being due to the fluid cooling medium. Therefore, another packaging and cooling method was devised in which the heat-producing electronics are mounted on a cold-plate through which a cooling fluid is pumped. The difference in these packaging concepts in terms of volume and weight is indicated below:

Controller/Inverter Rating (Amps)	Evaporative Cooling Vol (in^3)	Evaporative Cooling Wt (lbs)	Cold-Plate Cooling Vol (in^3)	Cold-Plate Cooling Wt (lbs)
50	172	11.5	56	4.0
100	426	22.5	113	7.2
150	508	36.0	169	11.1
200	672	48.0	225	14.3

Although the volume and weight saving with the cold plate cooling concept is impressive, some of these savings must go back into the liquid cooling system required to support this concept. The liquid cooling system is described in paragraph 4.9.4.

Configuration studies were continued after completion of Phase II and have resulted in the following actuation system changes which are reflected in Tables 19, 20, and 21:

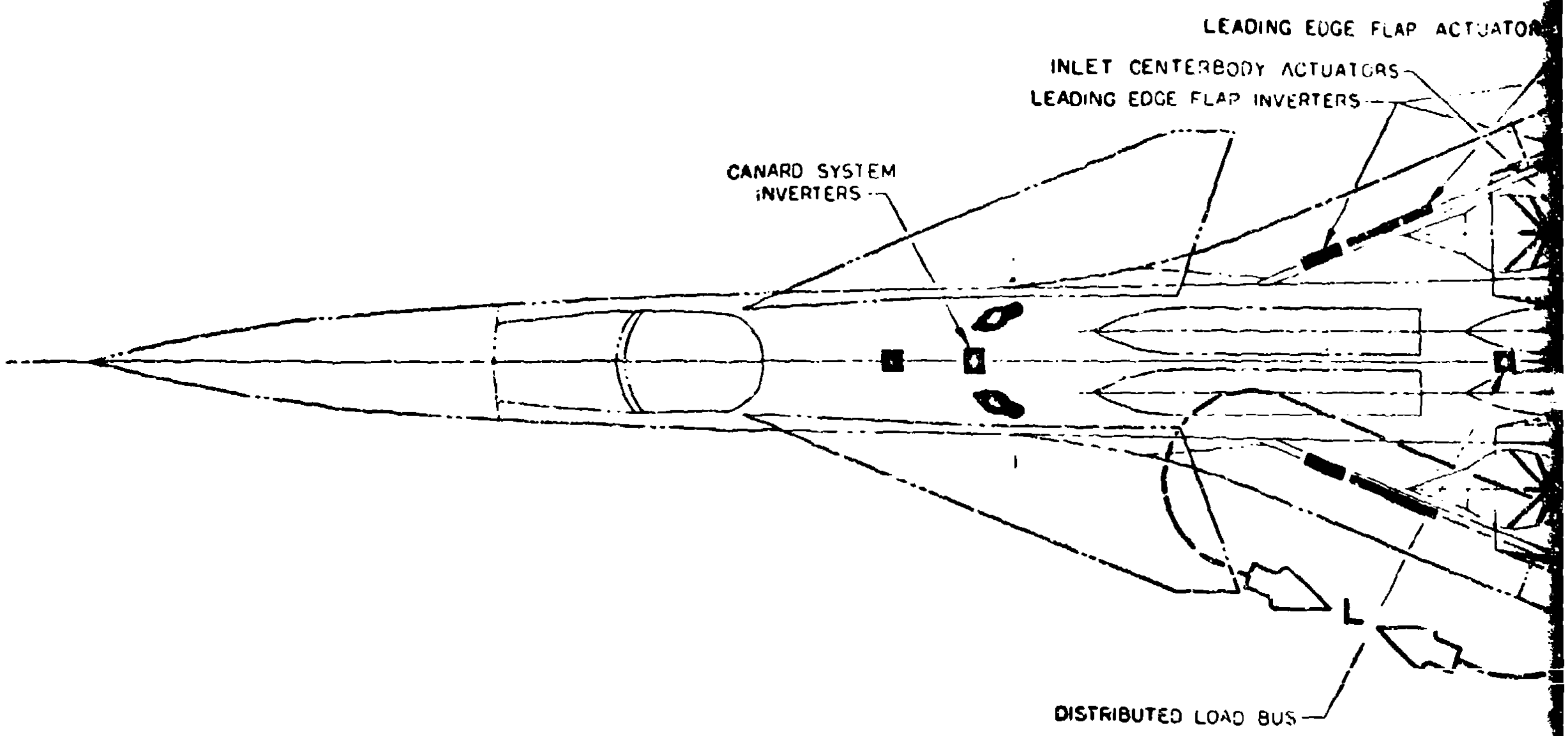
TABLE 20 ALL-ELECTRIC AIRPLANE ACTUATION SUMMARY - LANDING GEAR

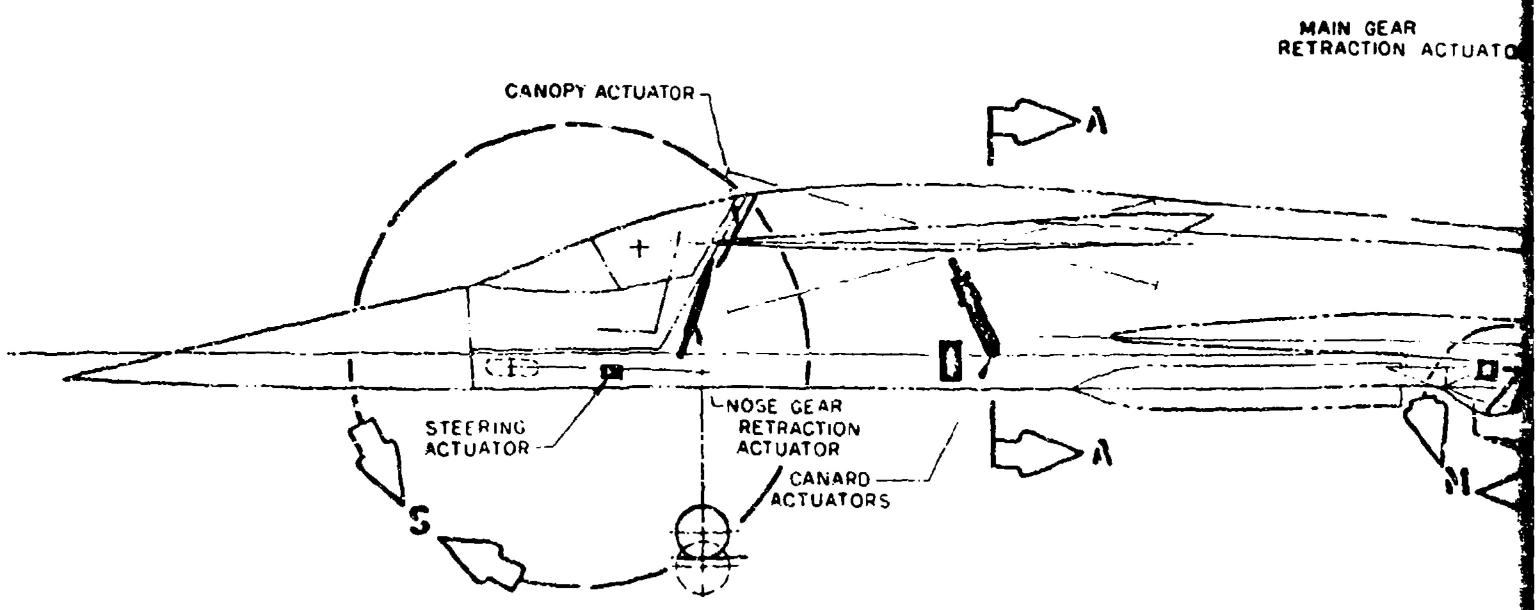
Actuator Function	Actuator Type	Peak Motor HP	Controller/ Inverter	Controller/Inverter Cooling
Main Gear Retraction	Linear Ballscrew & 270V DC Motor	7	Yes	Convection
Nose Gear Retraction	Linear Ballscrew & 270V DC Motor	7	Yes	Convection
Nose Gear Steering	Rotary EMA & 28V DC Motor	0.75	*	Convection
Main Gear Brakes	8 PM Motors, Ring Gear & Ballscrew Ram Per Wheel	0.33 per Motor	*	Convection

* Combined braking and steering control

TABLE 21 ALL-ELECTRIC AIRPLANE ACTUATION SUMMARY - MISCELLANEOUS

Actuator Function	Actuator Type	Rated Load	Controller/Inverter Cooling Required	Controller/Inverter Cooling Method
Aerial Refueling Door Actuator	Rotary EMA (28V DC Motor)	0.5 HP	No	-
Aerial Refueling Nozzle Latch Actuator	Linear EMA (28V DC Motor)	1750 lbs	No	-
Canopy Actuator	Linear EMA (28V DC Motor)	0.5 HP	No	-
Gun Drive	270V DC Motor (20,000 rpm) and gearbox	25 HP	Yes	Cold Plate
ECS Boost Compressor	270V DC Motor	50 HP	Yes	Cold Plate
ECS Pack Compressor	270V DC Motor	11 HP	Yes	Cold Plate
ECS Fan	270V DC Motor	43 HP	Yes	Cold Plate
Emergency Generator	Driven by IPU	-	-	-


LEADING EDGE FLAP ACTUATOR
INLET CENTERBODY ACTUATORS
LEADING EDGE FLAP INVERTERS
CANARD SYSTEM INVERTERS
L
DISTRIBUTED LOAD BUS


MAIN GEAR RETRACTION ACTUATO
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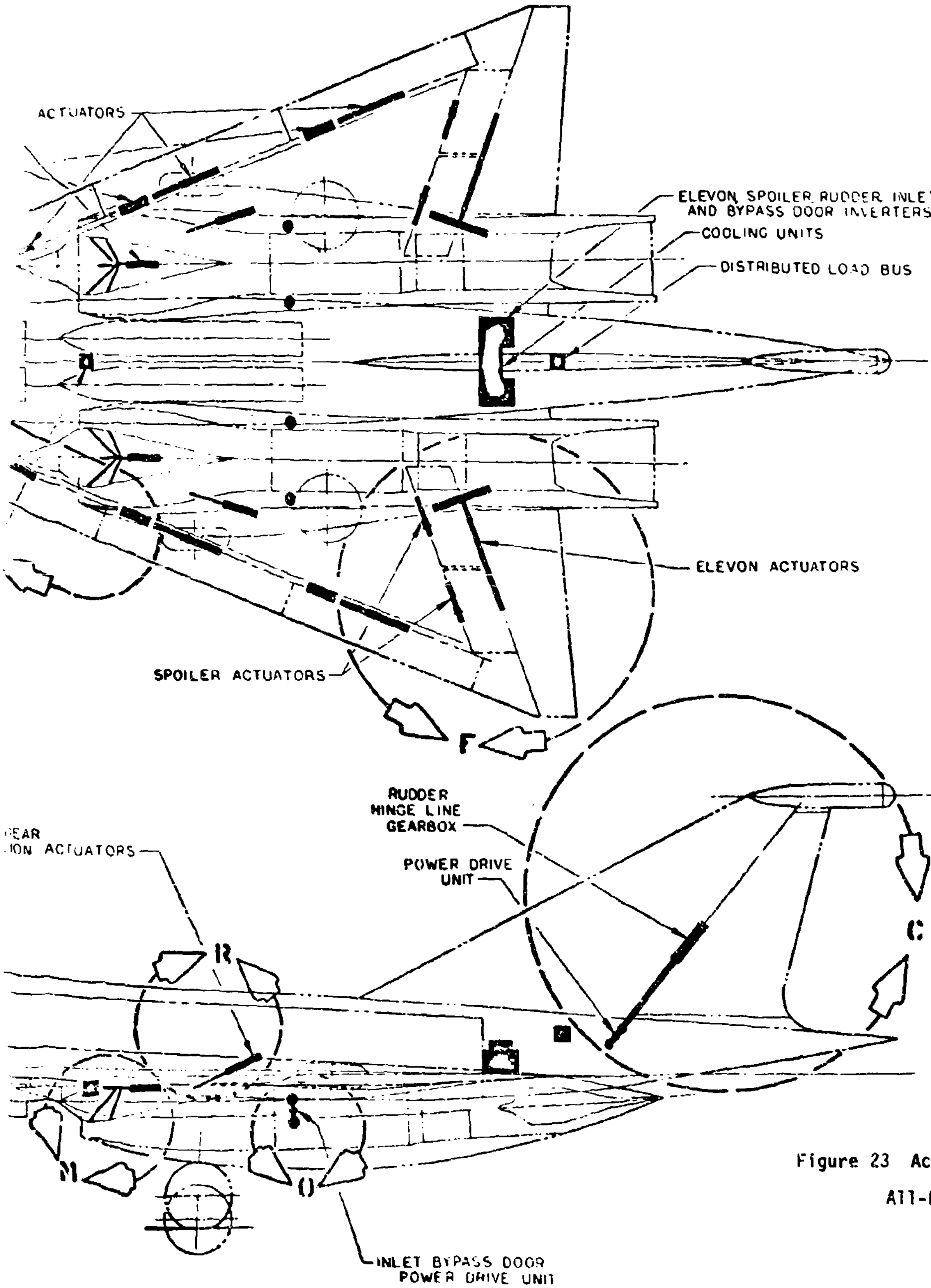
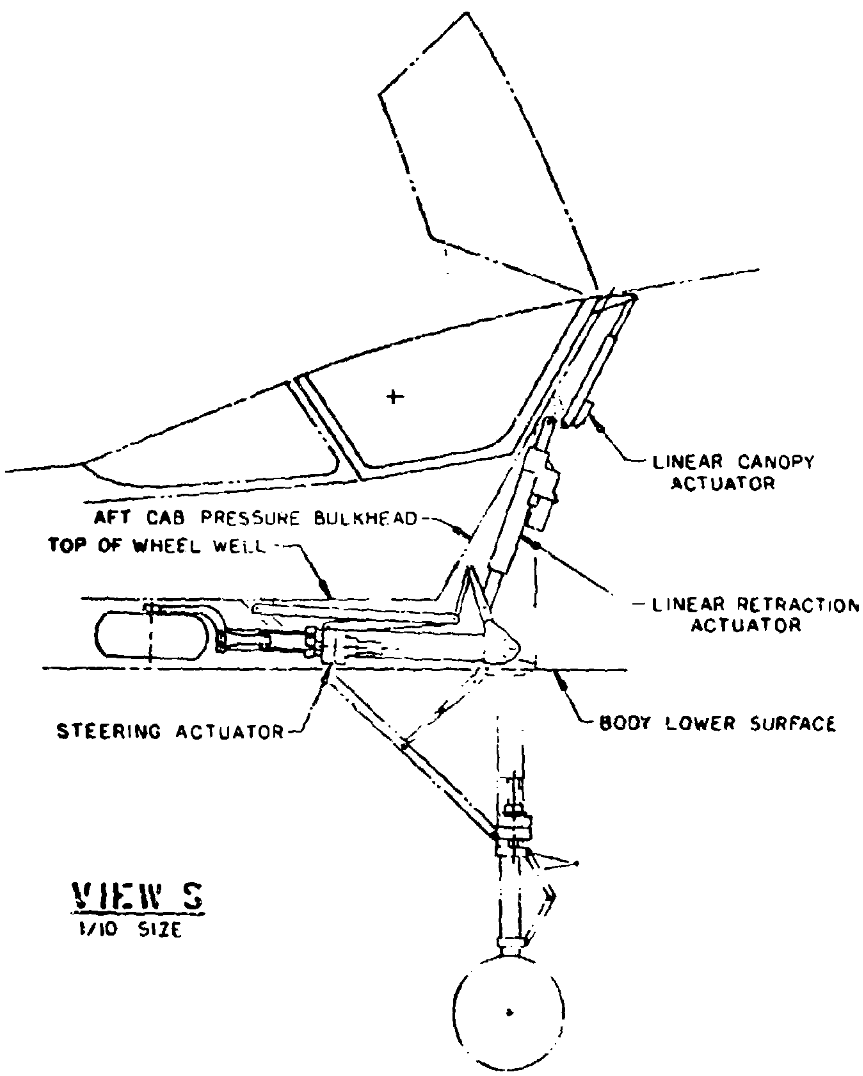


Figure 23 Actuation Systems Location

All-Electric Airplane



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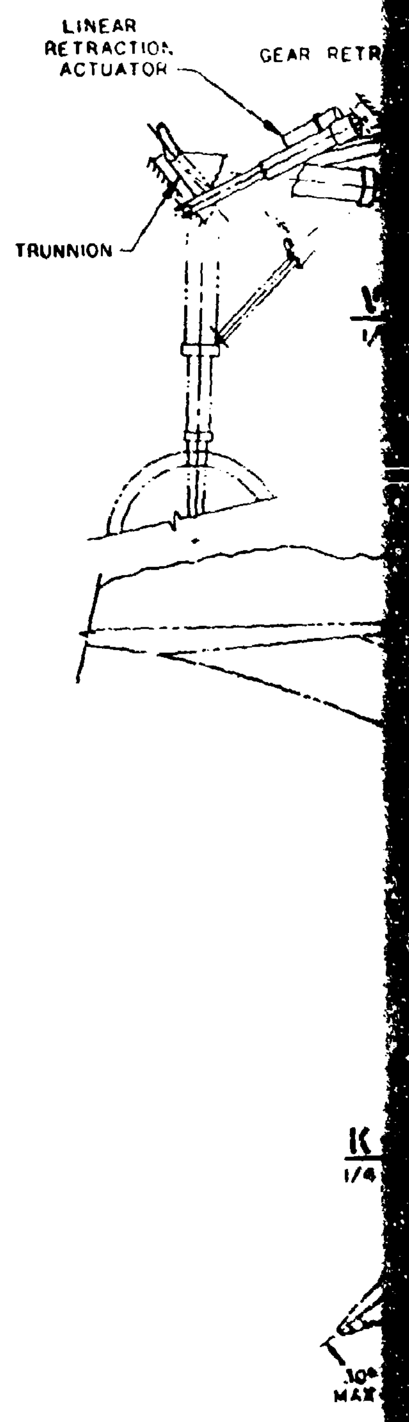
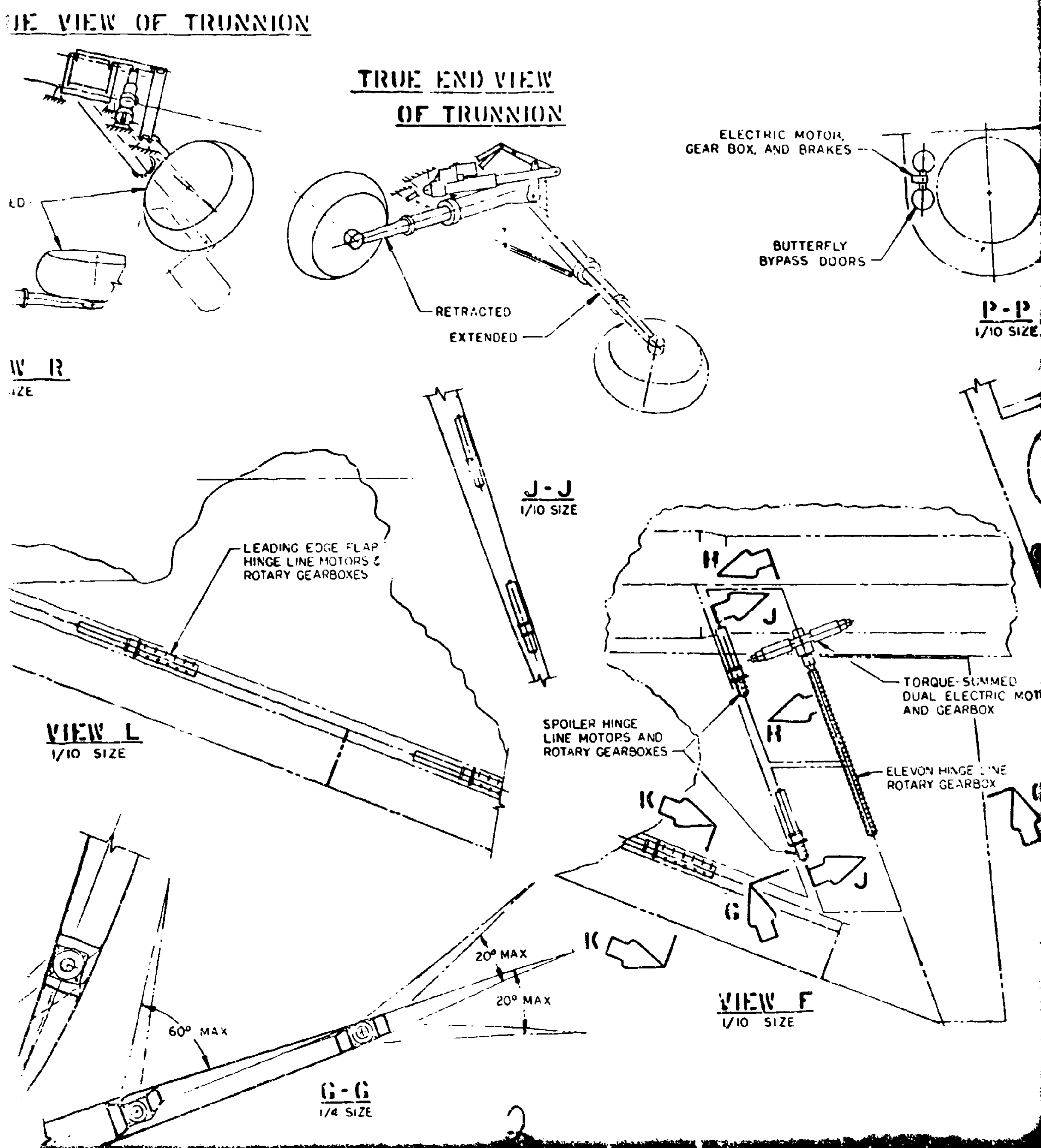
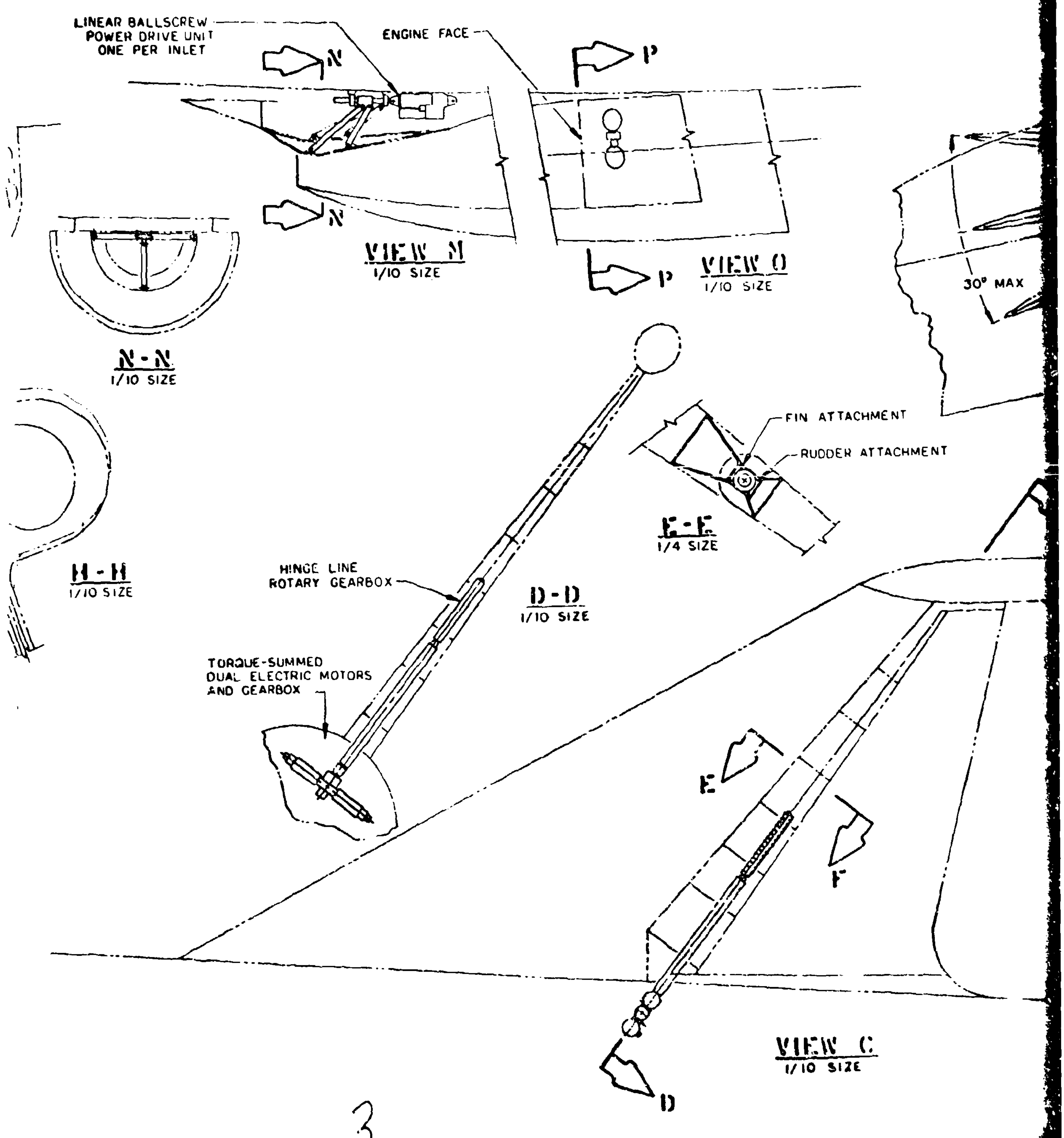




VIEW OF TRUNNION
TRUE END VIEW OF TRUNNION
ELECTRIC MOTOR, GEAR BOX, AND BRAKES
BUTTERFLY BYPASS DOORS
RETRACTED
EXTENDED
P - P
1/10 SIZE
J - J
1/10 SIZE
LEADING EDGE FLAP HINGE LINE MOTORS & ROTARY GEARBOXES
VIEW L
1/10 SIZE
H
J
TORQUE-SUMMED DUAL ELECTRIC MOT AND GEARBOX
SPOILER HINGE LINE MOTORS AND ROTARY GEARBOXES
ELEVON HINGE LINE ROTARY GEARBOX
K
G
20° MAX
20° MAX
60° MAX
VIEW F
1/10 SIZE
G - G
1/4 SIZE



LINEAR BALLSCREW
POWER DRIVE UNIT
ONE PER INLET
ENGINE FACE
N
N
P
P
VIEW M
1/10 SIZE
VIEW O
1/10 SIZE
30° MAX
N - N
1/10 SIZE
FIN ATTACHMENT
RUDDER ATTACHMENT
E - E
1/4 SIZE
H - H
1/10 SIZE
HINGE LINE
ROTARY GEARBOX
D - D
1/10 SIZE
TORQUE-SUMMED
DUAL ELECTRIC MOTORS
AND GEARBOX
E
F
D
VIEW C
1/10 SIZE

3

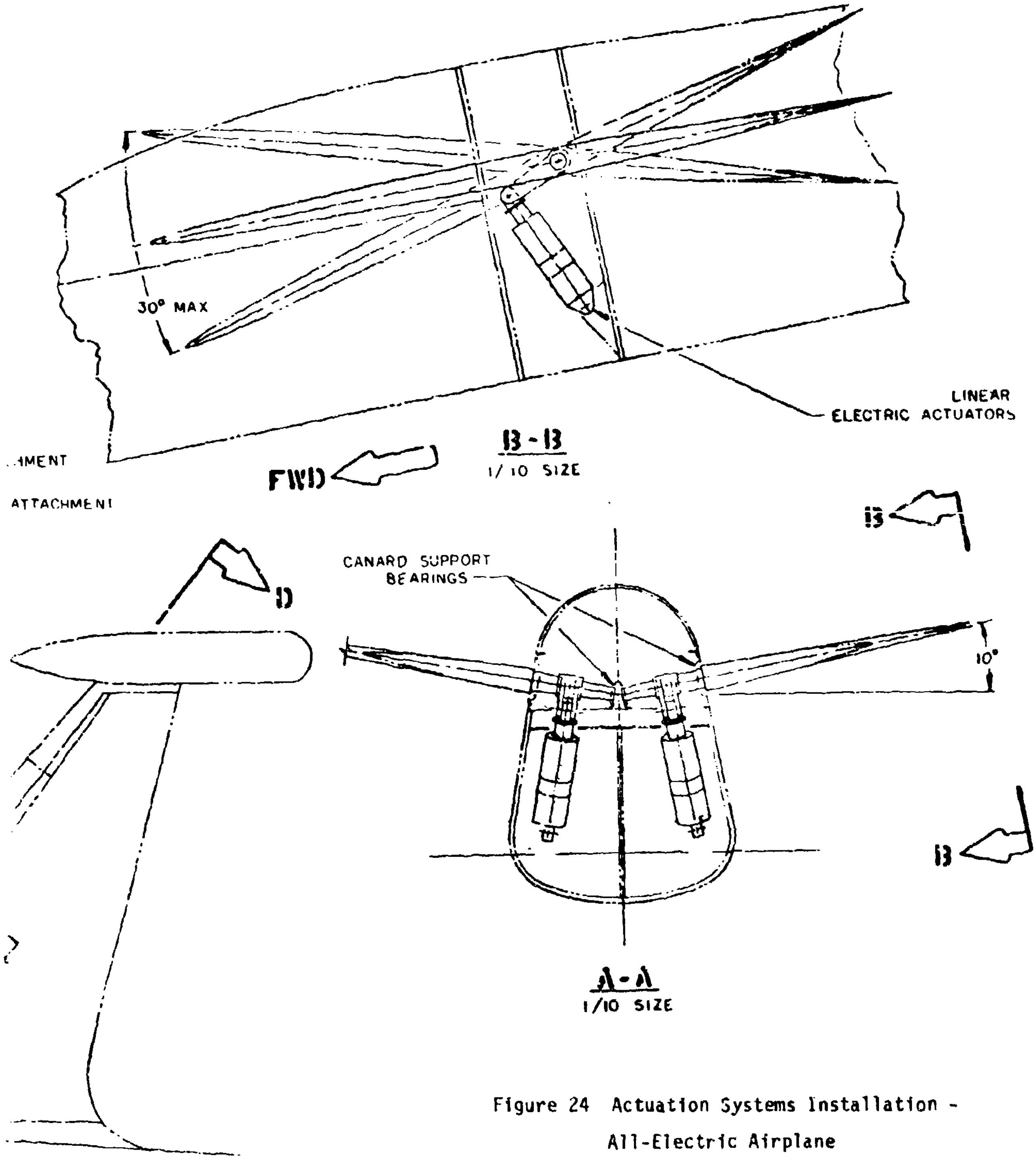


Figure 24 Actuation Systems Installation - All-Electric Airplane

Canard -	changed cooling of controller/inverter from shared E/E to cold plate
Rudder -	changed from IAP to EMA system
LE Flaps -	changed cooling of controller/inverter from forced air to cold plate
Landing Gear Retraction -	changed from AC motors to 270V DC motors (both main and nose gear)
Aerial Refueling and Canopy -	changed from AC motors to 28V DC motors (3 places)
Gun Drive -	added controller/inverter
ECS Boost Compressor -	changed cooling of controller/inverter from shared E/E to cold plate
ECS Pack Compressor and ECS Fan -	changed from AC motors to 270V DC motors and added controller/inverters with cold plate cooling

The rationale for these changes is covered in the following paragraphs which cover these functions.

4.3 Flight Control Actuation

4.3.1 Canard

Actuation trades considered the two canard surfaces interconnected as well as separated, as was done for the Baseline Airplane. The selected configuration is a ballscrew actuator driving each canard surface (not interconnected). The redundancy requirements as specified in Table 4 are met by using three motors, magnetically summed on the same shaft, to power each actuator. The motors are sized so that with one motor failed, the remaining two motors can power the actuator at rated load and speed. The configuration is shown in Figure 24 View A-A, and was selected for the following reasons:

(1) Significant weight sav ng over the other two types of EMA and the IAP configurations.

(2) Deleting the interconnection between the two canard surfaces saves weight and reduces complexity. The added actuation redundancy for separate surface control has minimal weight impact since no additional control capability is added in terms of increased power.

The actuation system for each of the two canard surfaces consists of the following components:

Ballscrew Actuator	38.0 pounds
270V DC Motor (3 required @ 8.0 lbs)	24.0 pounds
Controller/Inverter (3 required @ 7.7 lbs)	23.1 pounds
Total Weight per Surface	85.1 pounds

4.3.2 Elevons

Actuation trades considered a hingeline actuation system, a body-mounted power drive unit (PDU) and hingeline gearbox configuration and an IAP. The body-mounted PDU, consisting of two motors and a torque summed gearbox, along with a hingeline rotary gearbox shown in Figure 24 View F, is the selected configuration for the following reasons:

(1) Less weight than hingeline EMA and IAP configurations.
(2) It is the only configuration considered that fits within the available envelope.

The actuation system for each of the two elevon surfaces consists of the following components:

PDU/Hingeline Gearbox	70.0 pounds
270V DC Motor (2 required @ 13.7 lbs)	27.4 pounds
Controller/Inverter (2 required @ 24.5 lbs)	49.0 pounds
Total Weight per Surface	146.4 pounds

4.3.3 Rudder

Two EMA configurations and three IAP configurations were evaluated during

Phase II and the PDU/hingeline gearbox EMA system selected for the rudder function because it has the least weight and complexity.

The actuation system for the rudder consists of the following components:

PDU/Hingeline Gearbox	39.0 pounds
270V DC Motor (2 Required @ 10.5 lbs)	21.0 pounds
Controller/Inverter (2 Required @ 14.0 lbs)	28.0 pounds
Total Weight, Rudder Actuation System	88.0 pounds

4.3.4 Spoilers

A single hingeline motor/gearbox for each spoiler segment was selected over other concepts for the following reasons:

(1) Lighter and simpler than other EMA concepts (e.g., PDU in body driving hingeline gearbox through a torque tube; ballscrew linear actuator)

(2) IAP offers no significant advantage over EMA actuation system

(3) A neat, compact installation is possible as shown in Figure 24 View F.

The actuation system for each of the four spoiler surfaces consists of the following components:

PDU/Hingeline Gearbox	10.0 pounds
270V DC Motor	5.0 pounds
Controller/Inverter	7.0 pounds
Total Weight per Spoiler	22.0 pounds

4.3.5 Leading-Edge Flaps

A single hingeline motor/gearbox for each leading-edge flap segment was the selected configuration for the same reasons as listed for the spoiler application, paragraph 4.3.4. Synchronization of the flaps is accomplished electrically.

The actuation system for each of the six leading-edge flap segments consists of the following components:

Hingeline Gearbox	34.7 pounds
270V DC Motor	6.5 pounds
Controller/Inverter	8.5 pounds
Total, per flap segment	49.7 pounds

4.4 Engine Inlet Control Actuation

4.4.1 Engine Inlet Centerbody

Only linear actuation concepts were considered since the centerbody geometry and operational requirements dictate the use of a linear actuator. The configuration selected is a linear ballscrew electromechanical actuator shown in Figure 24 View M.

The actuation system for each of the two engine inlet centerbodies consists of the following components:

Ballscrew Actuator	32.0 pounds
270V DC Motor	5.0 pounds
Controller/Inverter	7.5 pounds
Total weight, per engine	44.5 pounds

4.4.2 Engine Inlet Bypass Doors

The selected configuration, shown in Figure 24 View P-P, consists of one EMA (single motor plus planetary gearbox package) operating each pair of doors. The actuation system for each of the four pairs of bypass doors consists of the following components:

Planetary Gearbox	3.0 pounds
270V DC Motor	1.0 pound
Controller/Inverter	1.0 pound
Total Weight per Pair of Doors	5.0 pounds

4.5 Landing Gear and Brakes

4.5.1 Main Gear Retraction

The main gear retraction system consists of a linear ballscrew actuator powered by a 270V DC motor for each main landing gear. A separate controller/inverter is provided for each motor. This arrangement differs from the configuration selected during Phase II since it was powered by a 400 Hz AC motor. The weight difference is negligible, however, since the weight of the AC motor is nearly identical with the combined weight of the 270V DC motor and the controller/inverter. Installation of the main gear actuator is shown in Figure 24 View R.

The actuation system for each of the two main landing gears consists of the following components:

Ballscrew Actuator	20.0 pounds
270V DC Motor	5.0 pounds
Controller/Inverter	5.7 pounds
Total Weight per gear	30.7 pounds

4.5.2 Nose Gear Retraction

As in the case of the main gear retraction system, the configuration of the nose gear retraction system has changed from that selected during Phase II. The AC motor has been replaced by a 270V DC motor and a controller/inverter with a very slight decrease in weight. Installation is shown in Figure 24 View S.

The actuation system for the single nose landing gear consists of the following:

Ballscrew Actuator	20.0 pounds
270V DC Motor	5.0 pounds
Controller/Inverter	5.7 pounds
Total Weight, Nose Gear Actuation	30.7 pounds

4.5.3 Nose Gear Steering

The actuator configuration selected for nose gear steering is a rotary actuator powered by a 28V DC brush type motor. This configuration permits operation of the nose gear steering function during towing operations on the ground when the only source of power is the aircraft battery.

The actuation system for nose gear steering consists of the following components:

Rotary Actuator	20.0 pounds
28V DC Brush Type Motor	4.0 pounds
Total Weight	24.0 pounds

4.5.4 Main Gear Wheel Brakes

A study of electric brake actuation was made by Goodyear Aerospace Company.

Weight estimates for the selected wheel and brake are as follows:

Wheel Assembly	77 pounds
Brake Assembly	94 pounds

The brake actuation components have been segregated from the total brake assembly in order to permit a more meaningful comparison with the Baseline Airplane. The brake actuation system for each of the two main gears consists of the following components:

Bull Ring Assembly	7.0 pounds
Motor (8 required @ 0.75 lbs)	6.0 pounds
Total, per gear	13.0 pounds

4.6 Aerial Refueling System

The aerial refueling actuation system is similar to the hydraulically actuated system in the Baseline Airplane (paragraph 3.6) except that a rotary electromechanical actuator (EMA) is used for door actuation and a linear EMA is used for nozzle latch actuation. Rated loads and weights are as follows:

Door EMA (Rotary)

Rated Load	0.5 HP
Actuator Weight	8.0 pounds
Motor Weight	0.25 pounds
Total Weight	8.25 pounds

Nozzle Latch EMA (Linear)

Rated Load	1750 pounds
Actuator Weight	4.0 pounds
Motor Weight	0.7 pounds
Total Weight	4.7 pounds

Both actuators are powered by 28V DC brush type motors so that the system can be operated from battery power in an emergency.

4.7 Canopy Actuation

A linear EMA, with characteristics as listed below, was selected for canopy actuation:

Rated Load	0.5 hp
Actuator Weight	7.0 pounds
Motor Weight	1.0 pounds
Total Weight	8.0 pounds

The actuator is powered by a 28V DC brush type motor so that the canopy can be operated from battery power when other power sources are not available.

4.8 Gun Drive

The total power required for the 25-mm Gatling gun is 25 hp which includes 14 hp for the gun drive and 11 hp for the feed system. A 270V DC, 20,000 rpm, brushless motor was selected to provide the required power. Component weights are:

Gearbox	15.5 pounds
Motor	11.2 pounds
Controller/Inverter	9.8 pounds
Total Weight	36.5 pounds

4.9 Environmental Control System (ECS)

The ECS in the All-Electric Airplane is identical to that in the Baseline Airplane (Figure 13) except for the electrically driven components described in the following paragraphs.

4.9.1 ECS Boost Compressor

The ECS boost compressor is driven by a brushless DC motor with a weight of 21.4 pounds. The required motor controller/inverter weighs 19 pounds. Duty cycle is continuous during climb, cruise, and landing. No boost compression is required during flight at Mach 2.2 and 60,000 feet altitude.

4.9.2 ECS Pack Compressor

The ECS pack compressor compresses the fluid used by the refrigeration pack. It is driven by a brushless DC motor which weighs 11 pounds. The associated controller/inverter weighs 5 pounds and duty cycle is continuous.

4.9.3 Electronic Cooling Fan

The electronic cooling fan circulates air between the heat sink, provided by the ECS refrigeration pack, and the electronic equipment. It is a continuous duty unit driven by a brushless DC motor weighing 18.4 pounds and a controller/inverter at 16 pounds.

4.9.4 Liquid Cooling System

The actuation systems for the All-Electric Airplane described in paragraphs 4.2 through 4.9.3 include a total of 28 liquid-cooled controller/inverters. This paragraph describes the liquid cooling system needed to provide cooling for the controller/inverter.

Due to redundancy requirements in the flight control system, three separate cooling loops are required. Heat loads have been divided among the three loops as equally as possible and the components sized accordingly.

A schematic diagram of the system is shown in Figure 25 and component weights are summarized below:

Reservoirs (3)	9.9 pounds
Motor/Pump (3)	7.5 pounds
Controller/Inverter (3)	6.0 pounds
Heat Exchangers (3)	6.0 pounds
Tubing, fluid-total	22.1 pounds
Installation, wiring - total	30.0 pounds
Total Weight	81.5 pounds

4.10 Secondary Power System

The secondary power system for the All-Electric Airplane is the Electrical Power System.

4.10.1 Electrical Power System

The electrical power system was designed to meet the requirements of power quantity, power quality, and source redundancy for the power-by-wire flight control actuators and fly-by-wire control of those actuators, as well as the weapons systems, avionics, fuel control, and other utility systems that conventionally use electrical power. The generators also shall serve as motors for engine starting.

The objective in this phase of the study was to select the most competitive combination of electrical power generation and distribution system components that could be considered available in the 1990 plus time frame.

Before selecting the electrical system configuration, a comparison study was made to select the specific starter-generator and power conditioning equipment type to be used in the final trade study. Three basic concepts were

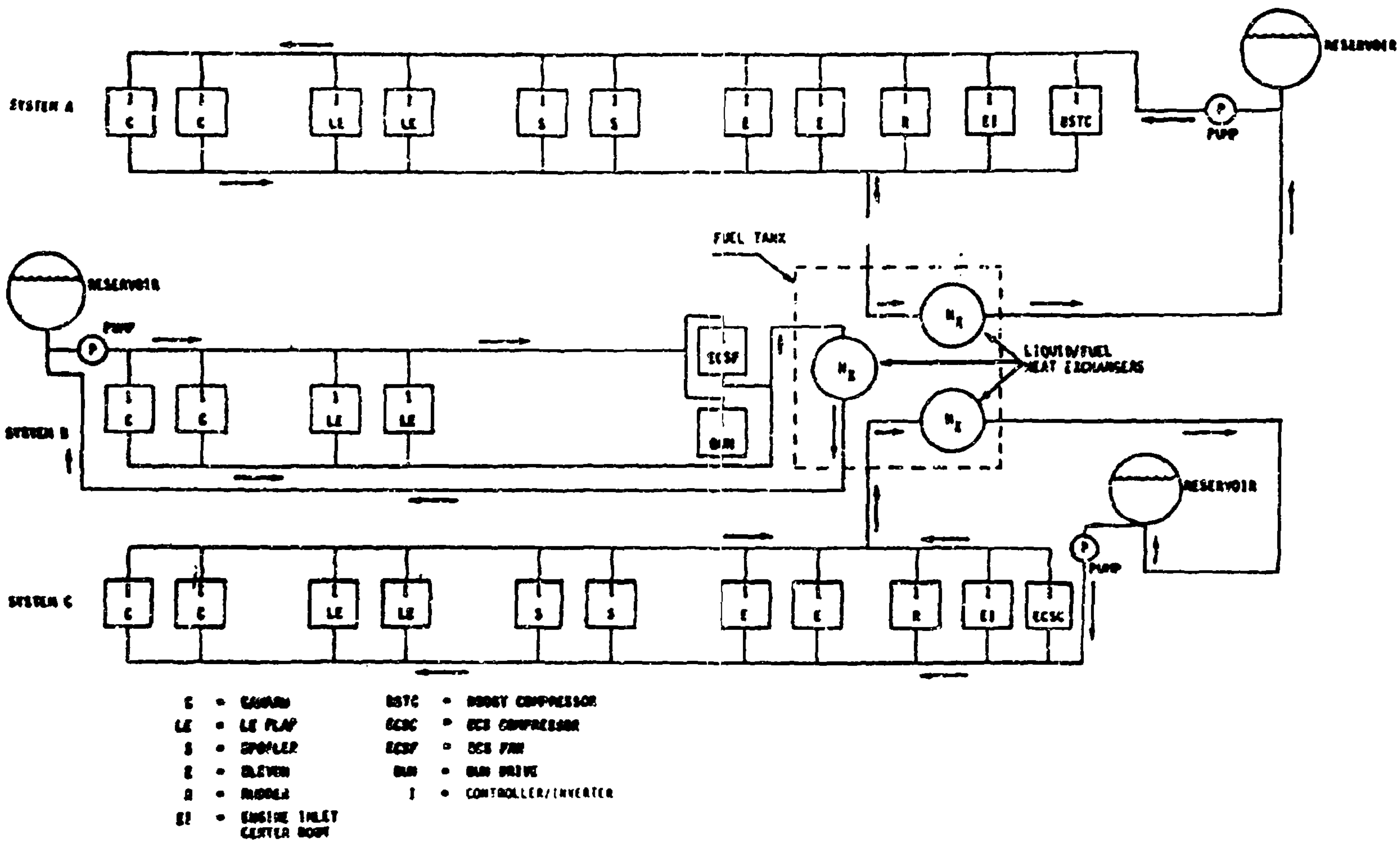


Figure 25 Liquid Cooling System — All-Electric Airplane

considered for processing the raw power (wild frequency, wild voltage) delivered by the generator:

1. Convert all of the power to regulated 120/208 volts, 400 Hz, and then rectify the desired portion to 270 volts DC.
2. Convert the desired portions of power from generated voltage and frequency directly to 270 volts DC and 120/208 volts 400 Hz.
3. Convert all of the raw power to regulated 270 volts DC and then invert the desired portion to 120/208 volts, 400 Hz.

The electrical power system configuration selected during Phase II is shown in Figure 26. This configuration met the electrical load profile shown in Figure 27. However, a major concern with this configuration was the relatively large weight of the cycloconverters (a total of 210 pounds for the 3 units). This, plus the fact that the rectifier bridges were lightly loaded, caused the question: Why can't loads be moved from 400 Hz AC to the DC busses, the cycloconverters eliminated, and the remaining AC requirements met by small inverters?

The electrical load analysis was examined and the following loads identified as those that could be powered by DC instead of 400 Hz power:

LOAD	CONNECTED LOAD (kW)
Primary Fuel Boost Pumps	7.3
Backup Fuel Boost Pumps	7.3
Fuel Transfer Pumps	7.3
Electronic Cooling Liquid Pump	2.0
Nose Gear Retract Actuator	5.5
Main Gear Retract Actuators	9.4
ECS Compressor Motor	9.4
ECS Fan Motor	37.6
Transformer - Rectifier Units	8.2
Lights	1.1
Aerial Refueling	0.4
Canopy Actuator	0.3
	95.8

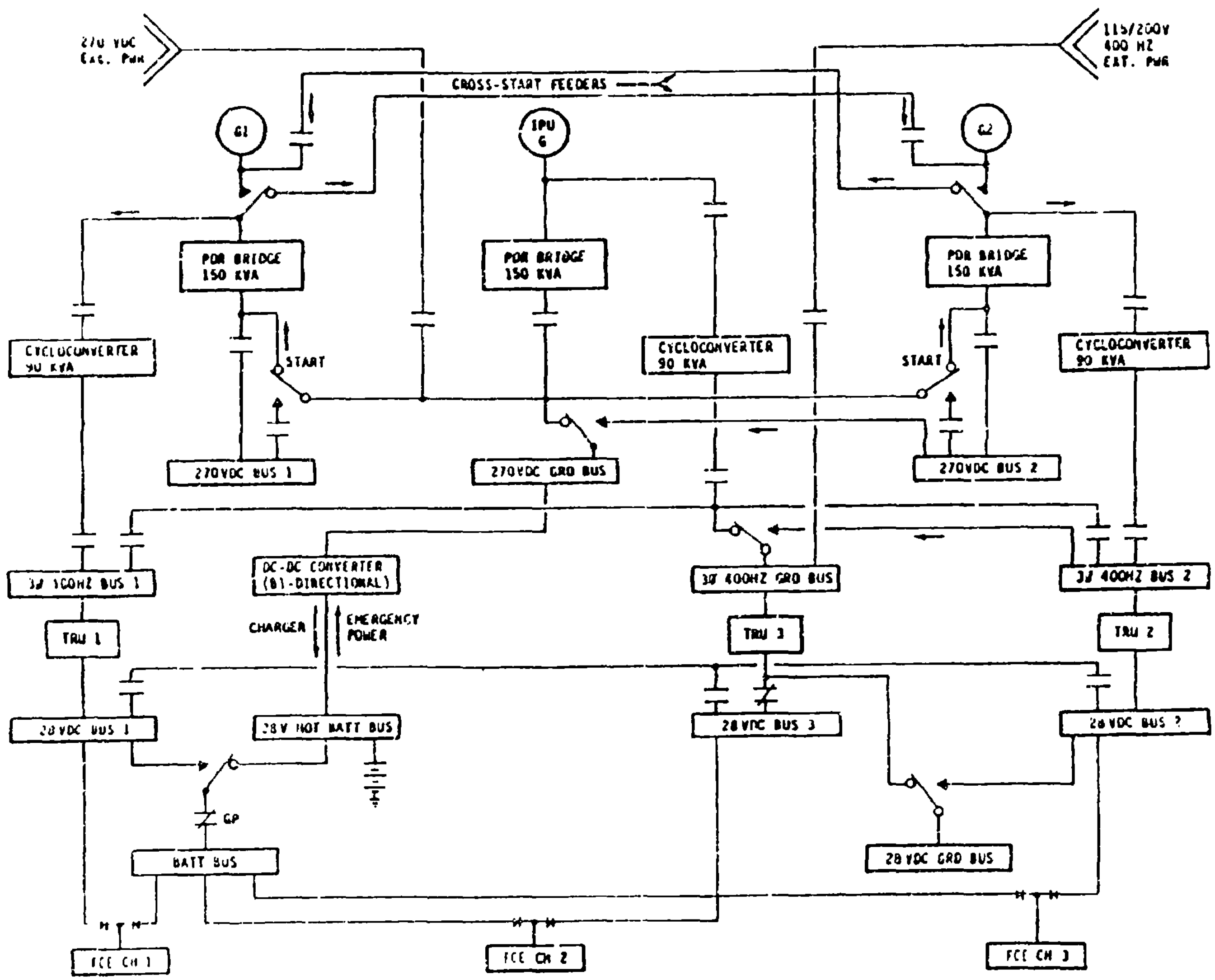


Figure 26 Electrical Power System - Phase II

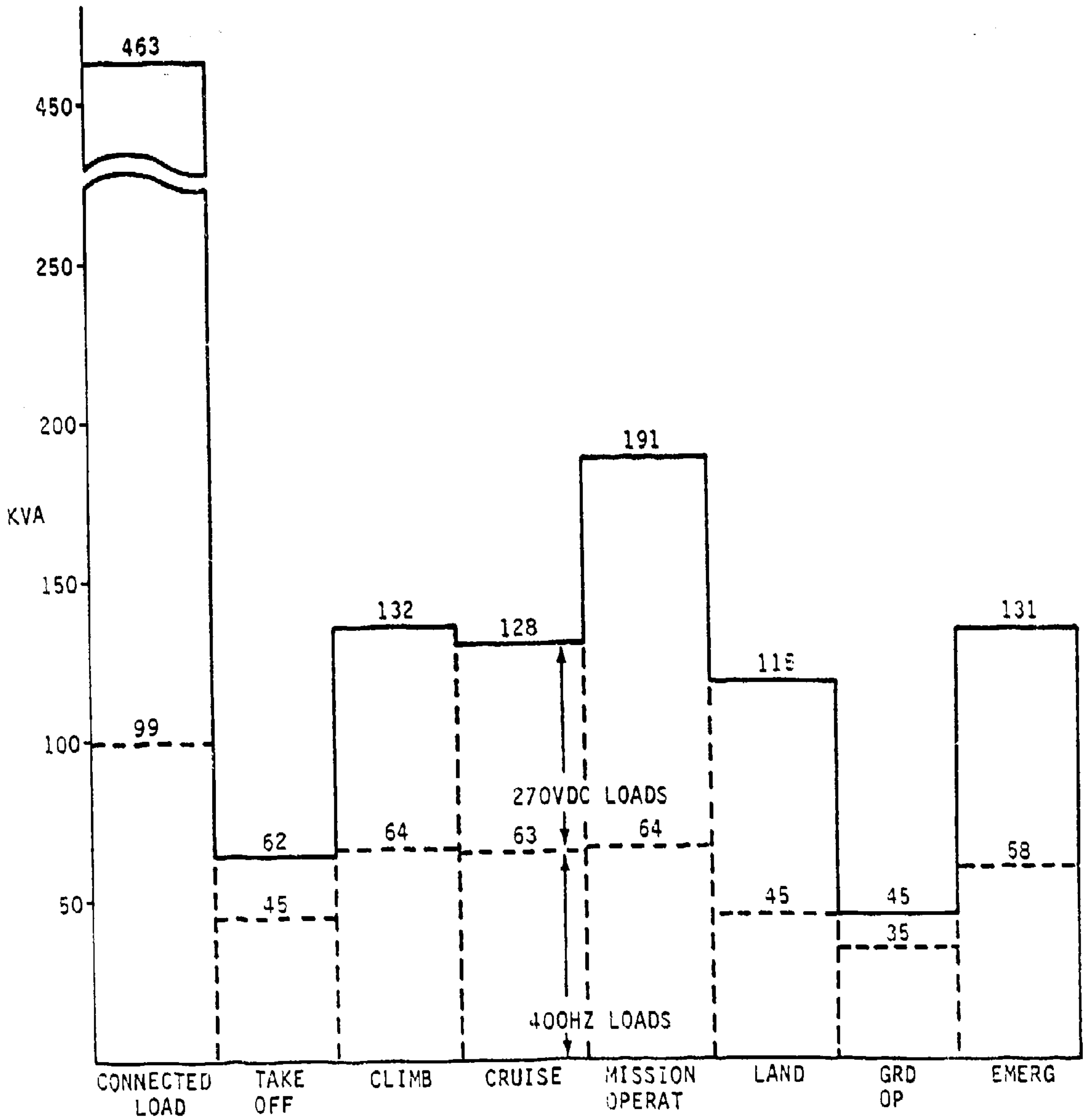


Figure 27 Electrical Load Profile - Phase II

Therefore, of the 99 kW of connected load supplied by 400 Hz AC in the Phase II configuration, all but 3.2 kW could be supplied by DC, either 270 or 28 volts.

These results encouraged further consideration of the power system change to the extent that the total electrical load analysis was revised (see paragraph 4.10.1.1), equipment changes identified, and estimates made of electrical system and cooling system impact. This led to the following conclusions:

1. The maximum continuous 400 Hz load requirement is 2.0 kW in the CRUISE flight condition.

2. The maximum continuous 28V DC load requirement is 2.9 kW in the TAKEOFF and LAND flight phases and less in other flight phases.

3. There is no significant change in total overall power requirement.

4. There is a reduction of 28 pounds in total equipment weight and a reduction of 142 pounds in major electrical power system components.

5. The effect on the liquid cooling system is a 5 pound weight increase.

The net weight saving of 165 pounds was sufficient reason for making this change in the electrical power system, but other considerations serve to reinforce this decision. First, the rectifier bridges are already of sufficient capacity to handle the additional 270V DC loads (they were sized by the engine starting requirement). Second, the rectifier bridges are more efficient and less complex than the cycloconverters, resulting in lower losses and increased reliability.

It was concluded that the configuration change was most desirable and therefore it was made, resulting in the schematic diagram shown in Figure 28. Power distribution for the actuation systems is shown in Figure 29.

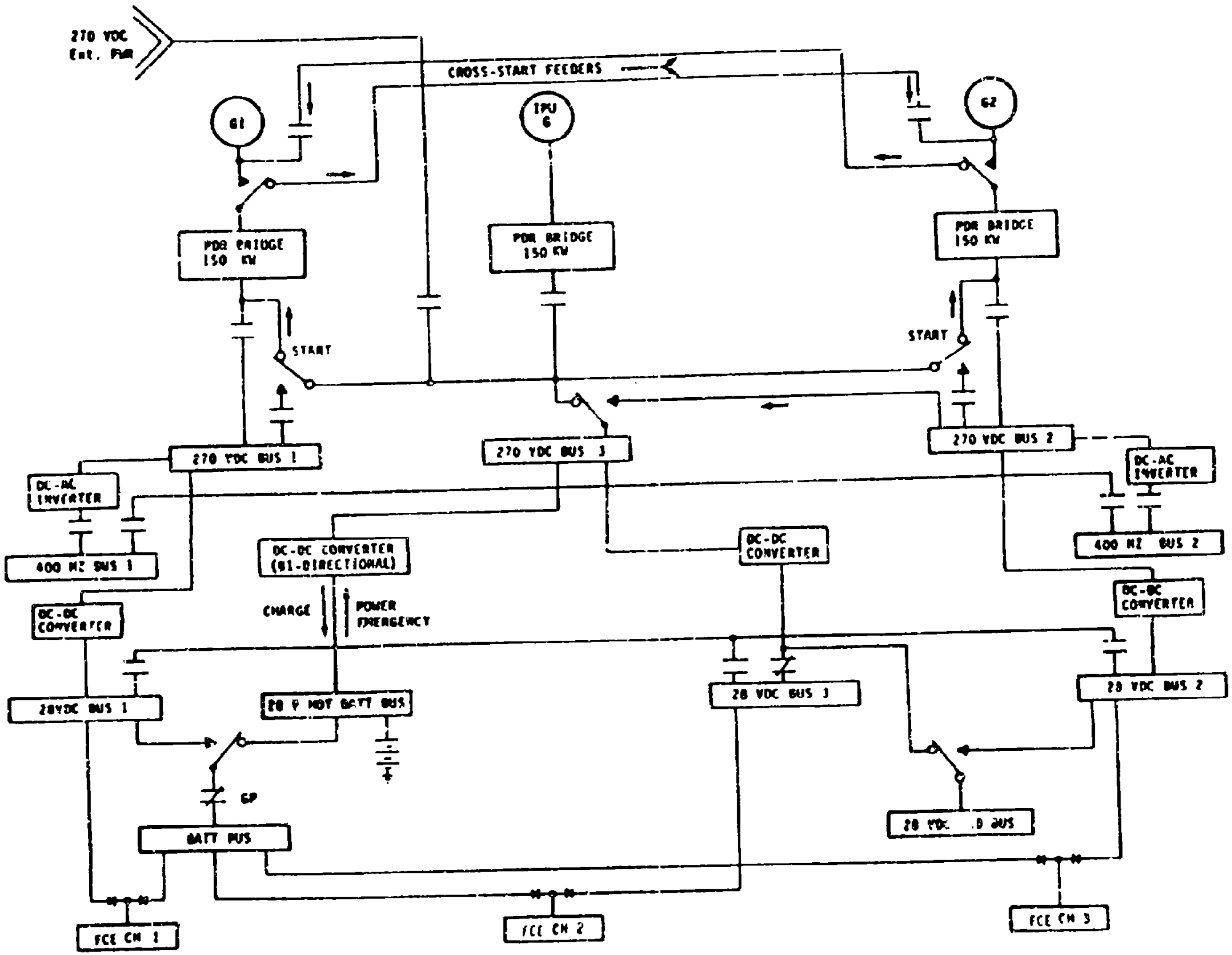


Figure 28 Electrical Power [illegible] - All-Electric Airplane

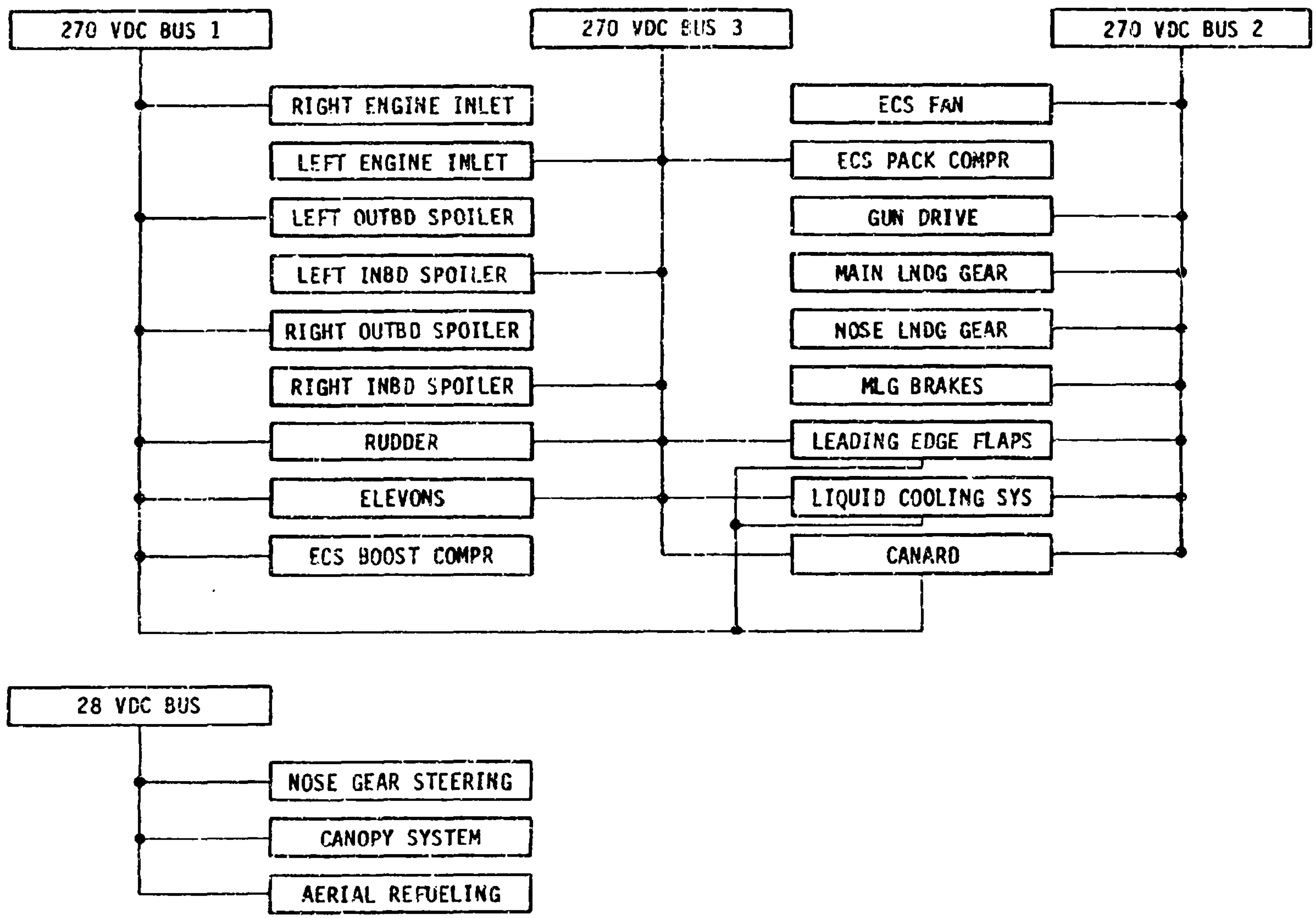


Figure 29 Electric Pow.. Distribution - Actuation Systems

4.10.1.1 Load Analysis

The updated electrical load analysis is shown in Figure 30 and Tables 22 through 24.

4.10.1.2 Selected System Arrangement

The two main generator/starters are mounted on the engine spinners as shown in Figure 31. An identical unit is mounted on the IPU power take-off pad. All other major components, listed in Table 25, are installed in the fuselage.

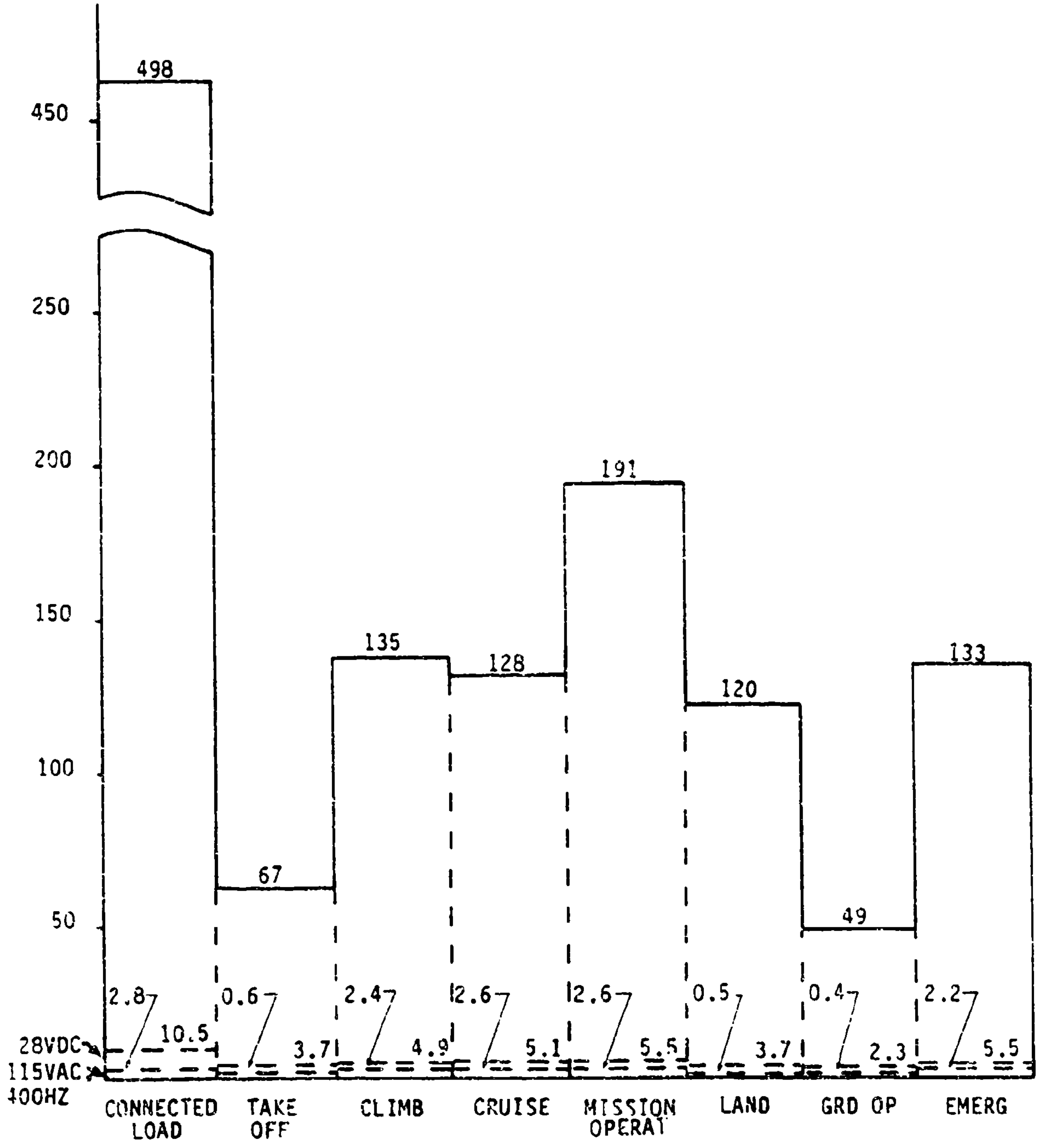


Figure 30 Electrical Load Profile - All-Electric Airplane

TABLE 22 ALL-ELECTRIC AIRPLANE LOAD ANALYSIS SUMMARY

SHEET 1 OF 1

ELECTRICAL LOAD TOTALS			COOL	MAX. CONNECTED LOADS, KW	UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, KW					KW, GROUND OPERATION	KW, SUSTAINED PEAK EMERGENCY	REMARKS, OPERATING TIMES
ITEM DESCRIPTION	NO	BUS			TAKE OFF	CLIMB	CRUISE	MISSION OPERAT.	LAND			
TOTAL 115 VAC POWER				2.80	0.55	2.43	2.53	2.55	0.49	0.43	2.24	
(TABLE 24 SHEET 5)												
TOTAL 28VDC POWER				8.16	3.18	2.48	2.57	2.93	3.20	1.88	3.23	
(TABLE 24 SHEET 5)												
TOTAL 115VAC + 28 VDC				10.47	3.73	4.91	5.12	5.48	3.69	2.31	5.47	
TOTAL 270 VDC POWER				487.82	63.51	129.72	122.94	185.47	116.17	46.94	127.63	
(TABLE 23 SHEET 3)												
TOTAL POWER				498.29	67.24	134.63	128.06	190.95	119.86	49.25	133.10	

TABLE 23 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 270 VDC LOADS SHEET 1 OF 3

AIRPLANE ACTUATORS ITEM DESCRIPTION	NO	BUS	COOL	MAX CONNECTED LOADS, KW 270 VDC	UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, KW: TAKE OFF 270 VDC	CLIMB 270 VDC	CRUISE 270 VDC	MISSION OPERAT. 270 VDC	LAND 270 VDC	KW. GROUND OPERATION 270 VDC	KW. SUSTAINED PEAK EMERGENCY 270 VDC	REMARKS, OPERATING TIMES
MAIN GEAR RETRACT ACTUATOR	2			9.40	0.48				0.48			
NOSE GEAR RETRACT ACTUATOR	1			5.50	0.28				0.28			
MAIN GEAR BRAKES	2			2.00					0.37	1.00		
ECS BOOST COMPRESSOR	1			44.32	5.70	44.32	44.32	0.88	44.32	5.70	44.32	
ECS FAN AND COMPRESSOR	1			45.00	27.00	45.00	45.00	45.00	35.00	27.00	45.00	
ELECTRONICS COOLING LIQUID PUMP	3			2.40	1.92	1.92	1.92	1.92	1.92	1.92	1.92	
* GUN FEED SYSTEM	1			9.65				0.40				5 SEC
* 25MM GUN DRIVE	1			12.29				0.51				5 SEC
* FLIGHT CONTROL ACTUATORS												
CANARD	3			58.51	0.70	1.41	0.70	19.13	2.81		2.81	
ELEVONS	2			108.24	2.72	8.18	8.18	61.82	8.18		8.18	
RUDDER	2			19.02	0.31	0.31	0.15	1.13	0.31		0.31	
SPOILERS	4			32.18		0.70		9.52	0.70		6.70	
LE FLAPS	6			59.97	0.71	7.06		14.41	7.06		7.06	
* ENGINE ACTUATORS												
CENTER BODY	4			25.80				2.51			2.51	
INLET BYPASS DOORS	8			0.50				0.05			0.05	
FUEL PUMPS												
PRIMARY BOOST	2			7.30	7.30	7.00	5.00	6.50	5.00	5.00	5.00	
BACKUP BOOST	2			7.30	6.00							
TRANSFER	2			7.30		5.00	5.00	5.00				
*Duty Factors have been applied		TOTALS		456.58	53.12	120.90	111.27	168.78	106.43	40.62	117.84	

TABLE 23 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 270 VDC LOADS

AIR VEHICLE AVIONICS			COOL	MAX CONNECTED LOADS, KW	UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, KW					GROUND OPERATION	KW. SUSTAINED PEAK EMERGENCY	REMARKS, OPERATING TIMES
					TAKE OFF	CLIMB	CRUISE	MISSION OPERAT	LAND			
ITEM DESCRIPTION	NO	BUS		270 VDC	270 VDC	270 VDC	270 VDC	270 VDC	270 VDC	270 VDC	270 VDC	
INTEGRATED INFORMATION MANAGEMENT SYSTEM--IIMS				5.40	5.40	5.40	5.40	5.40	5.40	5.40	5.40	
JTIDS/TACAN/SIF	1		YES	0.70	0.70	0.70	0.73	0.70	0.70	0.70	0.70	
GPS (GLOBAL POSITIONING SYSTEM)	1		YES	0.20	0.20	0.20	0.20	0.20			0.20	
MIRA (MULTI FUNCTION INERTIAL REFERENCE)	1		YES	0.20	0.20	0.20	0.20	0.20	0.20		0.20	
AIR DATA COMPUTER	1		NO	0.07	0.07	0.07	0.07	0.07	0.07	0.07	0.07	
* BATTERY CHARGING	1		NO	0.90	0.45	0.25	0.05	0.05	0.05	0.05		
AIR DATA SYSTEM HEATERS	1		NO	1.50	1.00				1.00		1.20	
TOTAL TEMPERATURE PROBE HEATER	1		NO	0.27	0.27				0.27			
BATTERY HEATER	1		NO	0.30	0.10		0.05	0.05	0.05	0.10		
WINDSHIELD HEATERS	1		NO	2.50	2.00	2.00			2.00		2.00	
RFSAR RADAR (SYNTHETIC APERTURE)	1		YES	5.00			5.00	5.00				
IRCM	1		YES	2.00				2.00				
ECM TRANSMITTER	2		YES	6.00				6.00				
WEAPONS HEATERS			NO	1.00				1.00				
TARGET ACQUISITION INFRAD RADAR			YES	1.50				1.50 TRANSIENT 0.5 SECOND SURGES				
GUN CONTROLS			NO	3.60								
TOTALS				31.14	10.39	8.82	11.67	22.17	9.74	6.32	9.77	

(* [illegible]) Factors have been applied

TABLE 23 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 270 VDC LOADS SHEET 3 OF 3

SUMMARY ITEM DESCRIPTION	NO	BUS	CODE	MAX. CONNECTED LOADS, KW 270 VDC	UTILIZATION F… ACTUAL APPLIED LOADS, KW: TAKE OFF VDC	CLIMB 270 VDC	CRUISE 270 VDC	MISSION OPERAT 270 VDC	LAND 270 VDC	KW, GROUND OPERATION 270 VDC	KW, SUSTAINED PEAK EMERGENT 270 VDC	REMARKS, OPERATING TIMES
LANDING GEAR ACTUATORS, SHEET 1				16.90	0.76				1.13	1.00		
ECS ACTUATORS, SHEET 1				91.72	34.62	91.21	91.74	47.80	81.24	34.62	91.24	
25MM GUN SYSTEM, SHEET 1				21.94				0.91				
FLIGHT CONTROL ACTUATORS, SHEET 1				277.92	4.44	17.66	5.03	100.53	19.06		19.36	
ENGINE ACTUATORS, SHEET 1				26.30				2.56			2.56	
FUEL PUMPS, SHEET 1				21.90	13.30	12.00	11.00	11.50	5.00	5.00	5.00	
AVIONICS, SHEET 2				31.14	10.39	8.82	11.67	22.17	9.74	6.32	9.77	
TOTAL 270VDC POWER				487.82	63.51	129.72	122.94	185.47	116.17	46.94	127.63	
*Duty factors have been applied		TOTALS										

TABLE 24 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 115 VAC AND 28 VDC LOADS

SHEET 1 OF 5

AIR VEHICLE AND AVIONICS LOADS			COOL	MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS GROUND OPERATION		WATTS SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT.		LAND						
ITEM DESCRIPTION	NO	BUS		115V 400 HZ	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	
*Relays, Contactors, Controls			No	300	750	250	500	250	500	200	450	200	500	200	500	150	300	150	80	
Instrumentation			?	100	150	100	150	100	150	100	150	100	150	100	150	100	150	100	30	Momentary Minor Load
*Refueling Controls			No		150															
*Fuel Transfer Controls			No		200		50		50		50		50		50				25	Momentary Minor Load
*Fuel Dump System			No		150															
*Fuel Tank Inerting Control			No		100								20							
*Engine Start Controls	2		No		150												150			
Fly-by-Wire Computer	4		Yes		760		260		260		260		260		260		260		260	
Displays & Controls	1		Yes																	
Navigation Lights	e		No		220		220		220		220		220		270				220	
Landing Lights	2		No		600		600								600				600	
Cockpit & Instrumentation Lights			No		250		200		200		200		200		200		200		150	
*Duty factors have been applied		TOTALS		400	2980	350	1980	350	1350	300	1330	300	1500	300	1980	250	1060	250	1365	

TABLE 24 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 115 VAC AND 28 VDC LOADS

AIR VEHICLE AND AVIONICS LOADS			COOL	MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS GROUND OPERATION		WATTS SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT.		LAND						
ITEM DESCRIPTION	NO	BUS		115V 400 HZ	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	115V 400 HZ.	28V. DC	
*Gyro System Heaters	2		No	100		50		50		50		50		50		50				
Windshield Wipers					200		200								200				200	
*ECS System Controls	2		No		150		30		30		30		30		30		30		30	
*Cabin Pressure Controls			No		100		20		20		20		20		20		20		20	
Oxygen System Controls			No		15						15		15						15	
Fire Detection, Warning & Extinguishing			No		25		25		25		25		25		25		25		25	
*Leading Edge Flaps Actuator Controls	2		No		15		5		5						5				5	
*Nose Gear Steering Controls	1		No		15		10								10				10	
*Anti-Skid System	1		No		15										10				10	
*Landing Gear Controls	3		No		30				5						5				5	
*Nose Gear Steering	1		No		540		45								45					9% Duty Cycle
*Aerial Refueling Door & Nozzle Latch Actuators	1				420						45								210	Latch Operates for 1 sec prior to door act. Door operates for 3 sec after latch completion
*Canopy Actuator	1				330												20		165	0.5 sec (On Grd)
*Duty factors have been applied		TOTALS		100	1855	50	335	50	85	50	135	50	90	50	350	50	95	0	695	

TABLE 24 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 115 VAC AND 28 VDC LOADS

SHEET 3 OF 5

CNI - COMMUNICATIONS NAVIGATION & IDENTIFICATION			COOL	MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS GROUND OPERATION		WATTS SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT.		LAND						
ITEM DESCRIPTION	NO	BUS		115V 400 HZ	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	
UHF/VHF Radio	1		Yes		50		50		50		50		50		50		50		50	
Faratune, Filter Cornett Clear	1		Yes		160		160		160		160		160		160		160		160	
Voice Processor	1		No	5		5		5		5		5		5		5		5		
KIT - 1A/T	1		Yes	37		37		37		37		37		37		37		37		
Recorder	1		Yes		240		240		240		240		240		240		240		240	
Wide Band Receiver	1		No		80		80		80		80		80		80		80		80	
MLS/ILS (Landing System)	1		No		20		20		20		20		20		20				20	
[illegible] factors have been applied		TOTALS		42	550	42	550	42	550	42	550	42	550	42	550	42	530	42	550	

TABLE 24 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 115 VAC AND 28 VDC LOADS

SHEET 4 OF 5

ARMAMENT/PAYLOAD LOADS			C O O L	MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS GROUND OPERATION		WATTS SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT.		LAND						
ITEM DESCRIPTION	NO	AUS		115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	
* Gun Controls	1		No		812							Transient 0.5 Sec. Surges								0.5 Sec. Surges
* Missile Launcher	1		No		448							Transient 0.5 Sec. Surges								0.5 Sec. Surges
*Expendables Dispenser	2		No		200								50							
Passive Sensors (RF & EO)	1		Yes	1500	300			1500	300	1500	300	1500	300					1500	300	
Warning Processor	1		Yes		Included in Above Load Data															
ECM 3-Rcv. Control	1		No		150								150							
Fire Control and Stores Management	1		No	150						150		150								
* Weapons Arming	1		No	50	50							Transient 2.0 Sec. Surges								2.0 Sec. Surges
* [illegible] Factors Have [illegible]	TOTALS			1700	1960			1500	300	1650	300	**1650	**500					1500	300	**[illegible] Transient Surges

TABLE 24 ALL-ELECTRIC AIRPLANE ELECTRICAL LOAD ANALYSIS OF 115 VAC AND 28 VDC LOADS

SUMMARY & TOTAL POWER				MAX. CONNECTED LOADS, WATTS		UTILIZATION CONSIDERATIONS - ACTUAL APPLIED LOADS, WATTS										WATTS GROUND OPERATION		WATTS SUSTAINED PEAK EMERGENCY		REMARKS, OPERATING TIMES
						TAKE OFF		CLIMB		CRUISE		MISSION OPERAT.		LAND						
ITEM DESCRIPTION	NO	BUS	COOL	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	115V 400 HZ.	28V DC	
Air Vehicle/Avionics Sheet 1				400	2980	350	1980	350	1380	300	1130	300	1506	300	1980	250	1060	250	1366	
Air Vehicle/Avionics Sheet 2				100	1855	50	335	50	85	50	135	50	90	50	350	50	95	0	695	
Communications, Navigation & Identification (CNI) Sheet 3				42	550	42	550	32	555	42	550	42	550	42	550	42	530	42	550	
Armament & Payload Sheet 4				1700	1960			1500	300	1650	300	** 1650	** 500					1506	300	** Plus heavy transient surge
AC Conversion Losses				540		110		486		511		511		98		86		448		
DC Conversion Losses					616		318		248		257		293		320		187		323	
Total AC Power				2802		552		2428		2553		2551		490		428		2240		
Total DC Power					8161		3183		2478		2572		2933		3200		1876		3233	
*Duty factors have been applied		TOTALS																		

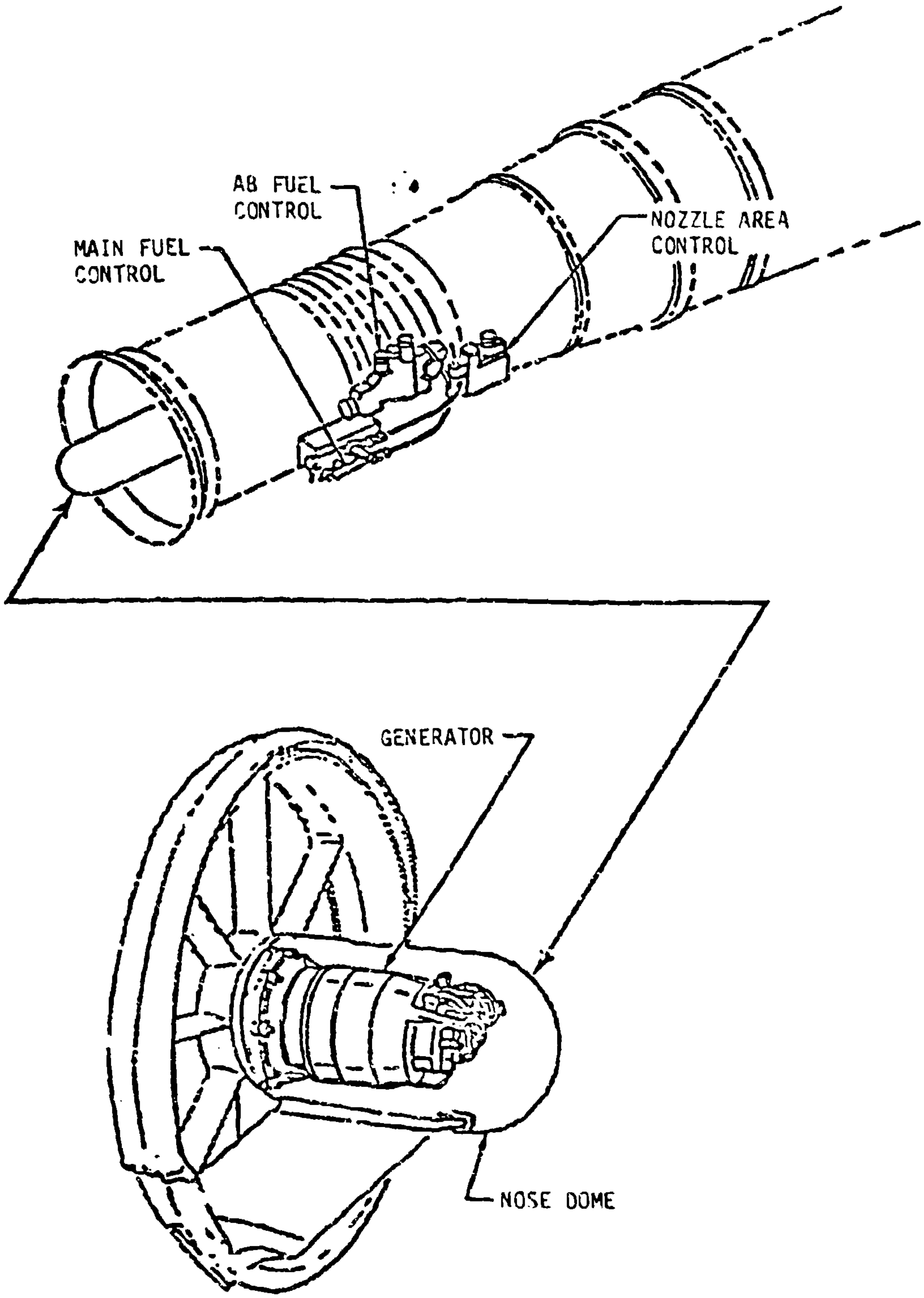


Figure 31 Single-Spool Engine Spinner-Mounted Generator/Starter

TABLE 25 ELECTRICAL POWER SYSTEM MAJOR COMPONENTS
ALL-ELECTRIC AIRPLANE

COMPONENT	NO. REQ'D	UNIT WEIGHT	TOTAL WEIGHT
Generator/Starter	3	.5	225
Phase Delay Rectifier Bridge	3	25	75
DC-DC Converter	4	17	68
DC-AC Inverter	2	34	68
Battery (2 @ 40 A-Hr)	2	75	150
AC Power Contactor 6PDT	2	18	36
AC Power Contactor 6PST	2	12	24
AC Power Contactor SPST	4	1	4
DC Power Contactor SPDT	3	9	27
DC Power Contactor SPST	6	6	36
Electrical Wiring and Connectors, total			231

TOTAL = 944 POUNDS

V TRADE STUDY

5.1 Trade Study Methodology

5.1.1 Approach

The trade study was conducted in accordance with the following outline:

a) Identify alternative airplane configurations to be evaluated.
b) Identify trade study ground rules.
c) Identify parameters to be considered in evaluation.
d) Assign weighting factors to each parameter.
e) Perform evaluation of alternatives.
f) Calculate weighted value totals for alternatives.

The parameters evaluated included:

- Weight
- Reliability and Maintainability
- Life Cycle Cost
- Performance
- Growth Potential
- Survivability
- EMC/Lightning Protection
- Environmental Constraints

Initially it was planned to assign weighting factors to each of these evaluation parameters by comparing each against every other parameter and judging which is the most important. However, this could not be done because the relative importance of each was dependent on many factors that were not a part of this study and different applications of a given equipment item on the same airplane could have a different relative importance. For example, weight may be the greatest single overriding factor in selecting a certain actuator for landing gear actuation whereas, survivability may be the most critical for a flight control function.

Therefore, the trades of each parameter were made between the alternative airplane configurations that were identified but the relative importance of each parameter was not assessed.

5.1.2 Ground Rules

The comparison of the Baseline and All Electric Airplane was made using the following ground rules.

It was assumed that all technological developments necessary to bring the various components and systems to the point where they would be ready for application to the study airplane would be completed by 1990 and the cost of these developments is not included in this trade study. The program for the development of this aircraft would begin with the release of a request for proposal in 1990 with an aircraft initial operational capability in mid to late 1990's. The airplane would have a service life of 10,000 flight hours and capability for 6,000 landings. The airplane would be designed for a 52 minute flight duration including takeoff, climb and cruise and 25 minutes for loiter, descent and landing. Both airplanes are assumed to be fly-by-wire.

The life cycle costs (LCC) were estimated for peacetime operation only using fiscal year 1981 dollars. The airplane would be operational for a period of 15 years, and utilize 288 flight hours per year. The airplanes would be grouped in squadrons of 24 units each. The LCC were computed for production quantities of 500 and 1000 units.

The LCC computations were done using the RCA PRICE Model and PRICE L Model. The LCC are computed based on the quantity of components, weight of components, amounts of structure, amounts of electronics (where applicable), complexity factors for engineering design, complexity factors for structure and electronics manufacturing, and density of electronics (where applicable). A detailed explanation of the RCA PRICE and PRICE L models is in Paragraph 5.4. Inputs are included in Appendix A so that the results achieved can be duplicated by a user. The RCA PRICE Model calculates the RDT&E, production cost, and creates the MTBF file for use in the RCA PRICE L Model where the operations and support costs are calculated for the LCC. The O&S cost includes mainly the supply (parts) and labor (maintenance) for the repair of an LRU. These costs are lower than would be achieved by a dedicated maintenance organization. In addition the LCC includes the cost only of the Baseline Airplane and the All-Electric Airplane. Crew, fuel, and all other systems normally included in a total aircraft LCC analysis are beyond the intended scope of this study and not included in this analysis.

5.2 Weight

The weight analysis of each airplane is limited to the actuation systems, secondary power systems, and the structural provisions to accommodate these systems. The other systems and components that are identical in each airplane, e.g., avionics, fuel, propulsion, etc., are eliminated from the analysis for simplicity.

Table 26 shows the weight summary for the two airplanes, Table 27 shows the weights for the actuation systems for the two airplanes and Table 28 shows the weights for the secondary power systems for the two airplanes. Source of the data in each case is shown on the table.

5.3 Reliability and Maintainability

Reliability Evaluation

An assessment of the reliability of both the Baseline and All-Electric Airplanes was conducted. Two parameters were used to compare the two airplanes. These were the probability of mission success and the probability of aircraft flight safety. These probabilities were computed as follows. The minimum equipment levels (MEL) for each subsystem for both mission completion and aircraft safety were defined and are summarized in Table 29. Fault trees were then constructed for both airplanes for loss of mission and loss of aircraft. These fault trees were developed down to the individual failure event that contributed to the top event. Certain failure contributing systems which were common to both airplanes were not considered in the computation since their effects would have the same effect on both airplanes. An example of this would be the FBW command signals since both airplanes were assumed to

TABLE 26

AIRPLANE WEIGHT SUMMARY

SYSTEM		BASELINE (LBS)	ALL-ELECTRIC (LBS)
Flight Control Actuators*		876	937
Engine Inlet Actuators*		58	109
Other Air Vehicle Actuators*		211	200
ECS*		21	172
Secondary Power System**		1202	944
	TOTAL	2367	2362

* From Table 27

** From Table 28

TABLE 27 WEIGHT SUMMARY -
ACTUATION SYSTEMS

FUNCTION	BASELINE	ALL-ELECTRIC
	WEIGHT-POUNDS	
FLIGHT CONTROL ACTUATORS	(875.8)	(937.2)
Canard	170.0	170.2
Elevons	300.0	292.8
Rudder	48.0	88.0
Spoilers	71.2	88.0
LE Flaps	231.6	298.2
Valves, total	14.0	--
Structural Weight Differential***	41.0	--
ENGINE INLET ACTUATORS	(58.0)	(109.0)
Centerbody	36.0	89.0
Bypass Doors	16.0	20.0
Valves, total	6.0	--
OTHER AIR VEHICLE ACTUATORS	(211.0)	(199.6)
Main Gear Retraction	37.8	61.4
Nose Gear Retraction	29.5	30.7
Nose Gear Steering	22.0	24.0
Main Gear Brakes	75.0*	26.0
Aerial Refueling	5.8	13.0
Canopy Actuator	3.9	8.0
Gun Drive	26.0	36.5
Valves, total	11.0	--
ECS	(21.0)	(172.3)
Boost Compressor	**	40.4
Pack Compressor	5.0	16.0
Electronics Cooling Fan	16.1	34.4
Liquid Cooling System	--	81.5

* Includes 6.0 pounds fluid

** Included in RH AMAD Gear Box

*** Due to Linear vs. Hingeline Actuation

TABLE 28

WEIGHT COMPARISON - SECONDARY POWER SYSTEM

ITEM	BASELINE (LBS)	ALL-ELECTRIC (LBS)
Hydraulic Power Generation	108	--
Hydraulic Power Reservoirs	22	--
Hydraulic Power Distribution	213	--
Hydraulic Fluid	99	--
AMAD Gear Boxes	231	--
Electric Power Generation	232	300
Electric Power Conversion	57	136
Electric Power Storage	75	150
Electric Power Distribution	165	358
	1202	944

Note: Baseline data from Tables 9, 12, and 18. All-Electric data from Table 25.

TABLE 29 SUMMARY OF MINIMUM EQUIPMENT LEVELS (MEL) SHEET 1 OF 3

AIRCRAFT EQUIPMENT	ACTUATORS			REMARKS
	QUANTITY PER AIRCRAFT	NEEDED FOR MISSION	NEEDED FOR SAFETY	
CANARD	3	2	1	TWO DUAL-TANDEM UNITS DEFINED AS THREE ACTUATORS IN THE BASELINE AIRPLANE
ELEVON RIGHT	2	2	1	LOSS OF EITHER ELEVON CONTROL IN HARD OVER MODE CAUSES LOSS OF A/C
ELEVON LEFT	2	2	1	
RUDDER	2	1	0	HARD OVER FAILURE CAN BE COMPENSATED FOR THRU USE OF ELEVONS
RIGHT L.E. FLAPS				
A	2	1	0	ANY TWO SURFACES ON ONE WING IN HARD OVER FAILURE MODE CAUSES A/C LOSS. FLAPS NEEDED FOR TAKE OFF.
B	2			
C	2			
LEFT L.E. FLAPS				
A	2	1	0	SAME AS ABOVE
B	2			
C	2			
RIGHT SPOILER				
INBOARD	1	0	0	ANY COMBINATION OF FAILURE MODES CAN BE COMPENSATED WITH OTHER SURFACES. NORMAL FAILURE MODE IS SURFACE-TRAILING WHICH DOES NOT CAUSE MISSION LOSS, HOWEVER, A SURFACE HARD OVER FAILURE CAUSES EXCESSIVE DRAG AND MISSION ABORT.
OUTBOARD	1	0	0	
LEFT SPOILER				
INBOARD	1	0	0	
OUTBOARD	1	0	0	

TABLE 29 SUMMARY OF MINIMUM EQUIPMENT LEVELS (MEL) SHEET 2 OF 3

AIRCRAFT EQUIPMENT	ACTUATORS			REMARKS
	QUANTITY PER AIRCRAFT	NEEDED FOR MISSION	NEEDED FOR SAFETY	
RIGHT ENGINE INLET	1	1	0	LOSS OF EITHER ENGINE INLET CONTROL ONLY AFFECTS ENGINE EFFICIENCY.
LEFT ENGINE INLET	1	1	0	
NOSE GEAR STEERING	1	1(a)	0	(a) NEEDED FOR TAXIING FOR TAKEOFF ONLY
NOSE LANDING GEAR	1	1(b)	1(a)	(b) AFTER TAKEOFF, FAILURE TO RETRACT CAUSES MISSION ABORT
MAIN LANDING GEAR	1	1(b)	1(c)	(c) FAILURE TO EXTEND CAUSES LOSS OF A/C DURING LANDING. LOCKOUT VALVE PREVENTS LOSS OF A/C DUE TO FAILURE DURING ENEMY CONTACT.
MLG BRAKES				
RIGHT	1	1(a)	1(d)	(d) FAILURE OF EITHER BRAKE DURING LANDING ROLL OUT CAUSES AIRCRAFT LOSS
LEFT	1	1(a)	1(d)	
GUN DRIVE	1	1	0(e)	(e) FAILURE DURING ENEMY CONTACT CAN LEAD TO A/C LOSS
AERIAL REFUEL	1	1(f)	0(g)	(f) REQUIRED IF MISSION INCLUDES REFUELING
				(g) FAILURE TO ACHIEVE REFUELING ASSUMED TO CAUSE ABORT TO NEAREST AIRFIELD, BUT NOT LOSS OF AIRCRAFT

TABLE 29 SUMMARY OF MINIMUM EQUIPMENT LEVELS (MEL) SHEET 3 OF 3

AIRCRAFT EQUIPMENT	ACTUATORS			REMARKS
	QUANTITY PER AIRCRAFT	NEEDED FOR MISSION	NEEDED FOR SAFETY	
CANOPY ACTUATOR SYSTEM	1	0	0	BECAUSE OF ASSUMED MANUAL BACKUPS THE CANOPY ACTUATOR SYSTEM IS CONSIDERED TO NOT BE MISSION OR SAFETY CRITICAL
ELECTRICAL POWER SYSTEM	3	2	1	BOTH CONFIGURATIONS HAVE COMPLETE CROSS-TIE CAPABILITY BETWEEN THE THREE MAIN BUSES
HYDRAULIC POWER SYSTEM	3	2	1	BASELINE AIRPLANE ONLY
ENVIRONMENTAL CONTROL SYSTEM	1	1	0(h)	SINCE BOTH CONFIGURATIONS ARE DEPENDENT ON ELECTRONICS IN THE FLY-BY-WIRE SYSTEM FOR FLIGHT CONTROL, THE ECS IS CONSIDERED TO BE MISSION CRITICAL. IT IS ASSUMED THAT IT IS NOT SAFETY CRITICAL BECAUSE THE EFFECTS OF LOSS OF ELECTRONICS COOLING ARE SLOW ENOUGH TO ALLOW ABORT OF MISSION WITHOUT LOSS OF A/C. (h) THE LIQUID-COOLING PORTION OF THE ALL-ELECTRIC ECS IS SAFETY CRITICAL. AT LEAST ONE OF THE TRIPLE REDUNDANT SYSTEMS ARE NEEDED FOR FLIGHT CONTROL STABILITY.

have FBW flight control systems. A typical set of fault trees are shown in Figure 32. The detail fault trees are included in Appendix A.

Failure rates used as inputs to the fault trees were derived from direct field experience data, supplier predictions and failure rate tables (such as RADC's Nonelectronic Parts Reliability Data, 1978) in that order of preference. When failure rates of equivalent components in Military or commercial transport aircraft were used, the failure rates were multiplied by a factor of two to convert then to the fighter failure rates.

The information thus obtained, was then entered as input data to a computer program, Simplified Computer Evaluation of Fault Trees (SCEFT), which computed the probabilities of loss of mission and loss of aircraft and thus provided the relative reliability figures for the Baseline and All-Electric Airplanes. These reliability figures are not a comprehensive set of numbers but are just to provide a relative measure for evaluating aircraft with the two types of actuation and secondary power systems.

The computed reliability figures are as follows:

	Mission Success	Aircraft Safety
Baseline Airplane	0.995608	0.999868
All-Electric Airplane	0.995289	0.999864

The computer printouts are included in Appendix A.

<u>Maintainability Evaluation</u>

An assessment of the maintainability of the actuation and secondary power system was also conducted for the Baseline and All-Electric Airplanes. This comparison was made on the basis of the Mean-Time-Between-Failure (MTBF) of the two airplanes' actuation and secondary power systems. The design of the airplanes was not sufficiently detailed to evaluate the maintenance-critical characteristics such as accessibility and mean-time-to-repair. The MTBFs of

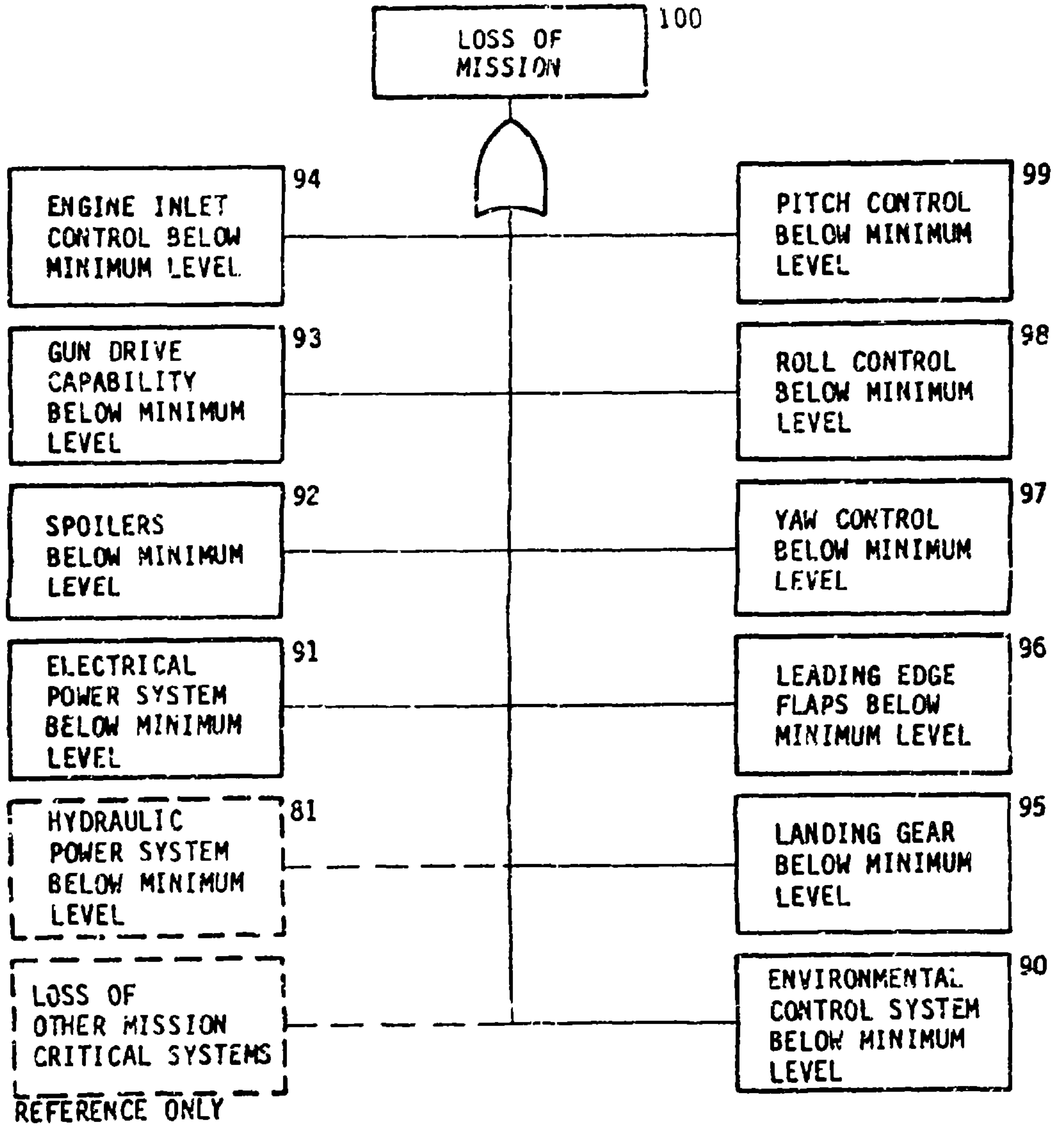


Figure 32 Sample Fault Trees (Sheet 1 of 3)

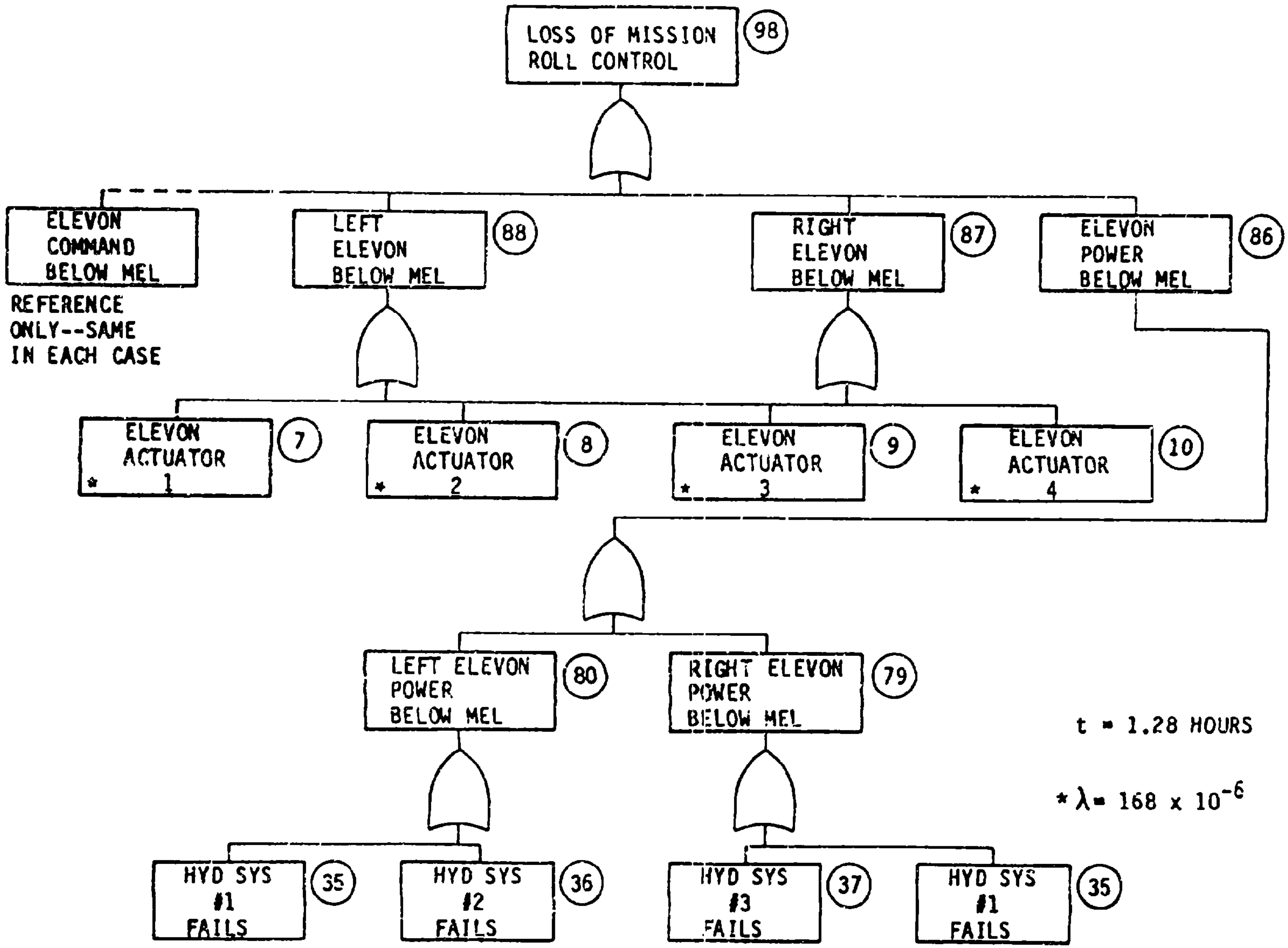


Figure 32 Sample Fault Trees (Sheet 2 of 3)

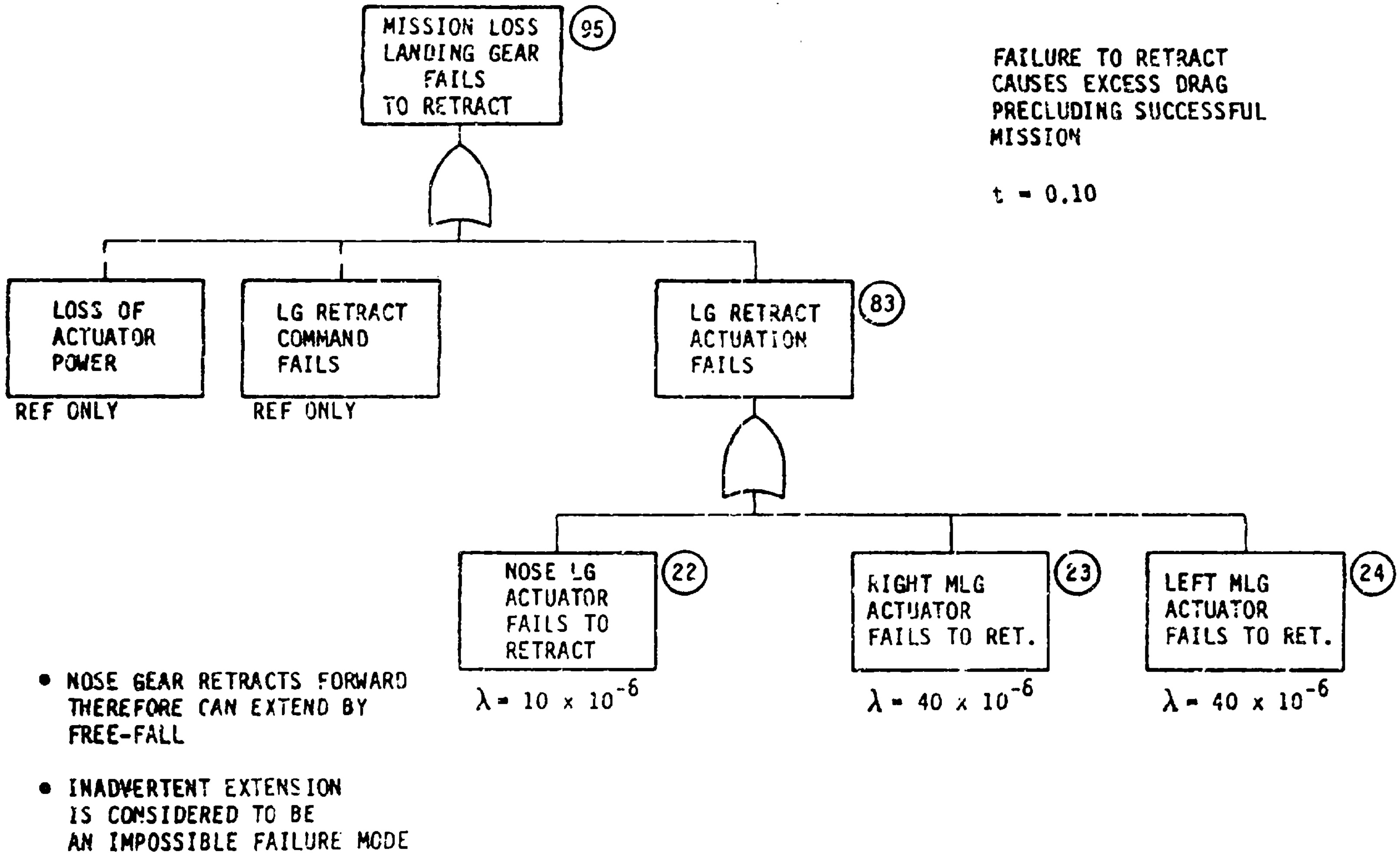


Figure 32 Sample Fault Trees (Sheet 3 of 3)

the components that were part of both airplane systems were computed by the RCA PRICE Program. The details of the input data used to obtain the MTBFs for the various components is discussed in Paragraph 5.4. The data obtained on the individual components was then combined to obtain the MTBF for the actuation systems and secondary power systems for the two airplanes as shown in Tables 30 and 31. The MTBF for the liquid loop system design to provide cooling to the EM actuation system controller/ inverters in the All-Electric Airplane was also computed and is shown separately in the table below. There was no comparable requirement in the Baseline Airplane.

MTBF IN FLYING HOURS

System	Baseline	All-Electric
Secondary Power	67	102
Actuation	139	53
Actuation Cooling	--	331
Overall MTBF	45	32

The ability of the airplanes to operate autonomously is enhanced by the capability to perform ground maintenance and system checks without having to run the engines. In the Baseline Airplane the power extraction from the engines is accomplished via the AMADs. This arrangement allows a ground check of the secondary power system via the IPU without having to run the engines. For the All-Electric Airplane the power extraction is accomplished via the engine spinner-mounted generators. Here the secondary power system checkout will not include the generators mounted on the engine spinners. However, due to the cross switching capabilities available in the electrical power system all the equipment downstream from the engine generators can be checked out via the IPU mounted generator. The generators selected for this airplane are permanent magnet generators. These generators contain no rotating rectifiers which are the components most likely to fail, thereby requiring the generators to be checked out. Therefore, the lack of the ability to operate the main generators without running the engines is not a serious drawback for the All-Electric Airplane.

TABLE 30 ACTUATION SYSTEM MTBF SUMMARY

SHEET 1 OF 2

FLIGHT CONTROL ACTUATION SYSTEM		MTBF (HRS) BASELINE	MTBF (HRS) ALL-ELECTRIC
CANARD		987	296
ELEVONS		2692	316
RUDDER		2613	921
SPOILER		2692	624
LE FLAPS		897	321
ENGINE INLET CENTERBODY		5385	1032
INLET BYPASS DOORS		2269	1369
	TOTALS	258	71

UTILITY ACTUATION SYSTEMS		MTBF (HRS) BASELINE	MTBF (HRS) ALL-ELECTRIC
LANDING GEAR EXT-RET		1705	809
NOSE GEAR STEERING		5390	3792
MAIN GEAR BRAKES		762	934
AERIAL REFUELING		3833	2994
CANOPY		5732	5359
	TOTALS	397	324

TABLE 30 ACTUATION SYSTEM MTBF SUMMARY
SHEET 2 OF 2

GUN AND ECS DRIVE		MTBF (HRS) BASELINE	MTBF (HRS) ALL-ELECTRIC
GUN DRIVE		2379	1949
ECS BOOST COMPRESSOR		INCL IN AMAD	1759
ECS PACK COMPRESSOR		8234	4302
ECS FAN		3578	2013
	TOTALS	1218	552
	TOTAL, ALL ACTUATION SYSTEMS	139	53

TABLE 31 SECONDARY POWER SYSTEM MTBF SUMMARY

BASELINE AIRPLANE

ITEM	MTBF (HRS)
ELECTRIC	
60 KVA GENERATOR	3276
60 KVA CYCLOCONVERTER	385
20 KVA GENERATOR	6204
TRANSFORMER-RECTIFIERS	1203
STATIC INVERTER	3344
40 AMPERE-HOUR BATTERY	43426
BATTERY CHARGER	5060
DISTRIBUTION AND WIRING	312
	132
HYDRAULIC	
33.5 HP MOTOR	5599
46 GPM PUMP	1239
RESERVOIRS	2998
HEAT EXCHANGERS	3739
FILTER MODULES	3590
CASE DRAIN FILTER MODULES	2089
TUBING, FITTINGS, VALVES	234
	151
POWER EXTRACTION EQUIPMENT	
LH AMAD GEARBOX	3421
RH AMAD GEARBOX	3358
ANGLE GEARBOX	4440
CLUTCHES-ANGLE GEARBOX TO AMAD - POWER TAKE-OFF	**
INPUT/OUTPUT DRIVE SHAFTING	**
STRUCTURAL PROVISIONS	**
	1226
TOTAL	67

** INCLUDED WITH GEARBOXES

ALL ELECTRIC AIRPLANE

ITEM	MTBF (HRS)
150 KVA STARTER/GENERATOR	1659
150 KW PDR CONVERTERS	595
2.1 KW DC - DC CONVERTER	546
2 KW STATIC INVERTERS	664
80 AH BATTERY	21713
DISTRIBUTION AND WIRING	225
TOTAL	102

5.4 Life Cycle Cost

Life Cycle Costs (LCC) were estimated for the actuation systems and secondary power systems for both the Baseline Airplane and the All-Electric Airplane, including costs for integration of the systems with each other. The LCC includes RDT&E, Production, Support Investment, and Operating and Support Costs. The LCC plan for this study is illustrated in Figure 33. Subsystem design was sufficient to estimate weights and volume of the individual Line Replaceable Units (LRU) for input to the cost model.

More detailed cost data would require preparation of procurement specifications to obtain detailed supplier cost estimates. In addition, it would require an increase in detail design to refine airplane provisions and installation details of the LRUs. This level of detail was considered to be beyond the scope of the Preliminary Design nature of this study and therefore the cost model was run at the LRU level.

The use of LCC, including operating and support costs, is the preferred approach for cost effectiveness analysis in this study. Essentially, it allows consideration of the trades between development and production costs, maintainability, reliability and survivability.

5.4.1 Cost Model

The LCC model used in the airplane actuation trade study was the RCA Program Review Of Information For Costing And Evaluation And Life Cycle Cost Model (PRICE L).

Some of the basic program ground rules for this study were as follows:

RCA - PRICE Cost Model
RCA - PRICE L Model
Prototype Hardware 10 Units
Prototype Spares 5 Units
Production Quantity 500, 1000
Flying Time 288 Hrs/Year

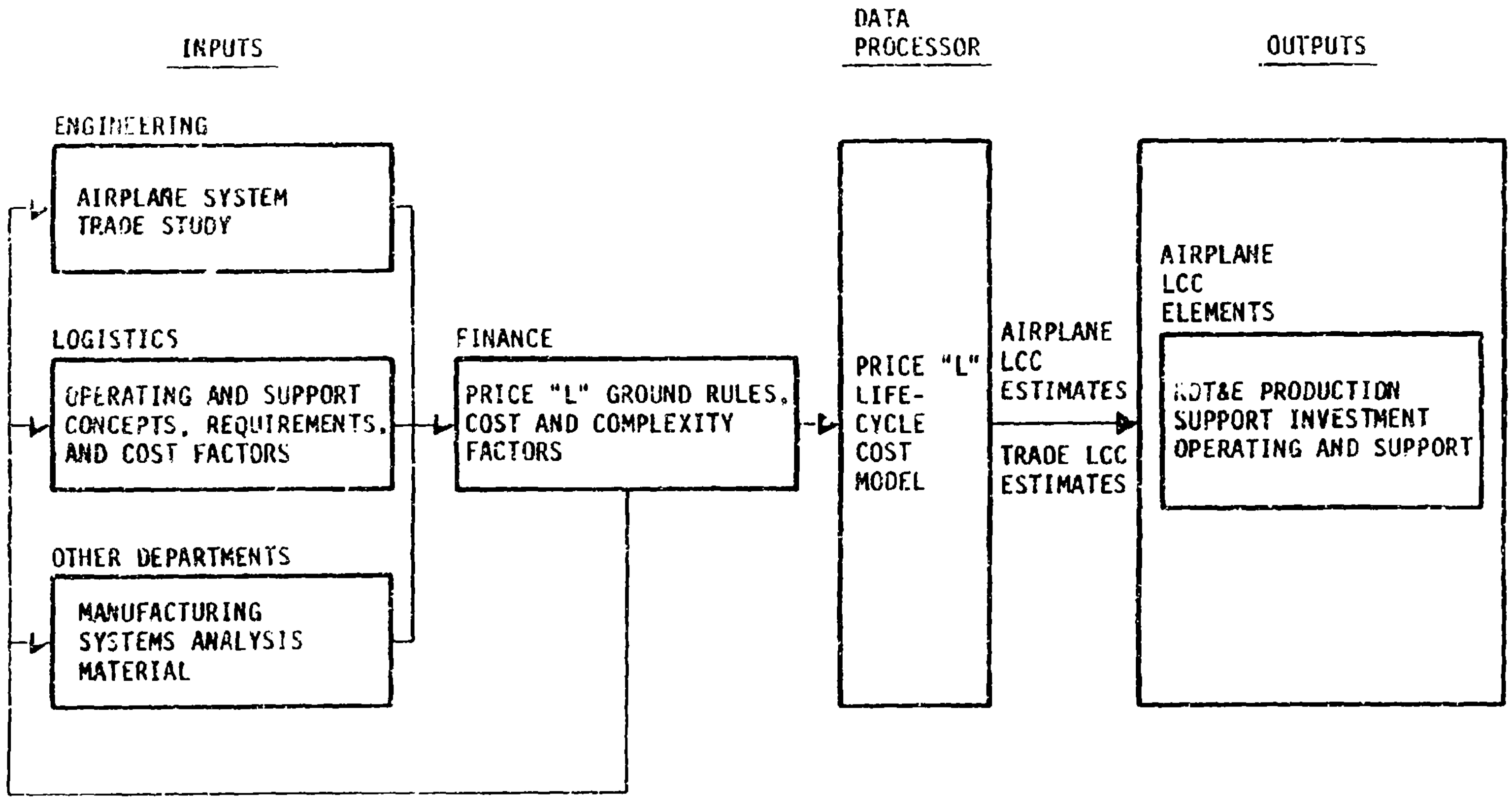


Figure 33 Life-Cycle Cost (LCC) Plan

Ground Operating Time Fraction 0.4
Operating Period 15 Yrs
Airplanes Per Squadron 24
All Costs 1981 $

Cost elements included in the model are described below.

5.4.2 RDT & E Costs

The development cost element in LCC includes those efforts required to develop previously undeveloped or partially developed components/systems. The study presupposes that the new technology items identified as requiring further development will have received the required development funding prior to the technology availability date (1990) of this airplane. Therefore, these costs are not included in the RDT and E. Involved are: (1) the research into what is required, what exists, how it will function, and how it will interact with the system; (2) the design which is the engineering required to mechanically configure components; and (3) the test and evaluation to see that performance meets the required specifications. Production non-recurring tooling and test equipment are part of this effort.

5.4.3 Production Costs

Production costs include the materials, labor, quality control, recurring, tooling, planning, and program management efforts required for making the components/systems for a given quantity buy. The production units may be produced inhouse (make) or procured outside (buy).

5.4.4 Support Investment Costs

Support investment costs include initial spares, ground support equipment, data, training, and other.

5.4.5 Operating and Support Costs

Operating and support costs include those efforts required to operate and maintain an airplane/system throughout its operational life. Maintenance support costs are significant costs and include the effort required to repair, rework, and replace parts at the operational level defined by the government.

5.4.6 Cost Estimating Technique

The PRICE L cost model has been used to estimate engineering development and manufacturing cost of electronic, electromechanical, hydraulic, and mechanical components. Numerous estimates using the PRICE L cost model were made to verify its accuracy. It was calibrated, when appropriate, with vendor quotes or by judgment based on historical data.

Support investment costs were estimated using the PRICE L cost model. These costs include support equipment and initial spares that were estimated based upon the logistic concept consistent with a 1990+ time frame.

Operating and support costs were also estimated using the PRICE L Model. PRICE L operates at the Line Replaceable Unit (LRU) level and provides an efficient method for developing operating and support costs at the time of hardware estimation. PRICE L allows evaluation of many logistic concepts in addition to reliability, maintainability, and weight.

Figure 34 presents a flow diagram of the inputs to the RCA PRICE Model and Figure 35 shows the development and production inputs required to calculate the cost of an LRU. A compilation of all the basic inputs that were used in the PRICE Model for this study is included in the Appendix. Figure 36 presents the inputs used for running the PRICE L Model. Table 32 presents the values calculated in the Operations and Support Cost portion of the PRICE L Model. Table 33 shows the LCC cost of a typical LRU in the study.

5.4.7 LCC Data

A summary of the LCC study results is presented in Table 34 and Figure 37. The LCC savings of the All-Electric Airplane over the Baseline are \$13M for 500 aircraft and \$23M for the 1,000 aircraft program.

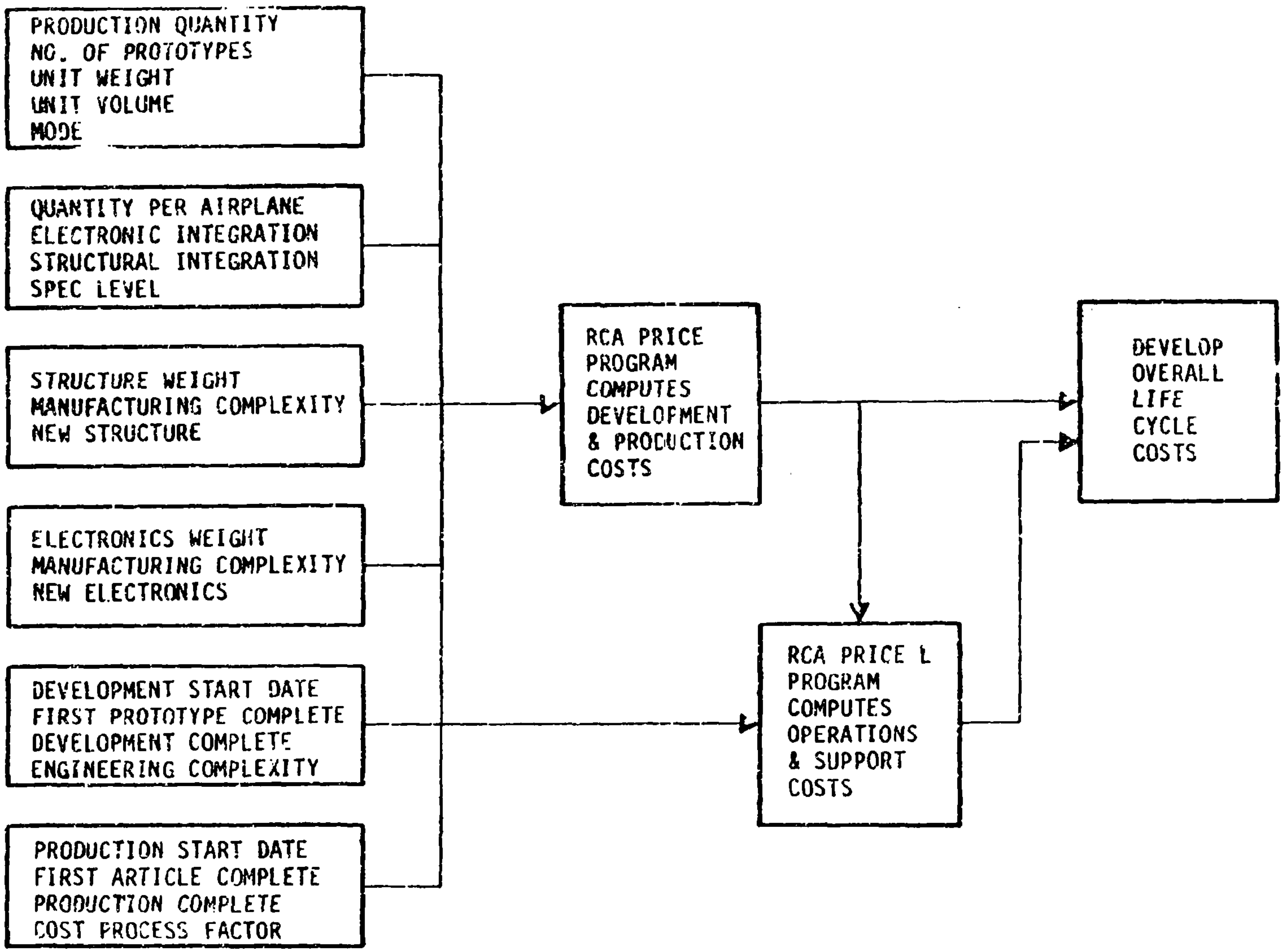


Figure 34 Life Cycle Cost Computation

Title: Power Drive Unit/Ball Screw Act. - Canard | Date:

General A	Production Quantity QTY 1000/2000	Prototypes PROTOS 30	Weight (lb) WT 38.0	Volume (ft^3) VOL .1652	MODE 2	
General B	Quantity/Next Higher assembly QTYNHA 2	NHA Integration Factors Electronic INTEGE 0	Structural INTEGS .5	Specification Level PLTFM 1.8	Year of Economics YRECON 1981	Year/Type of Technology YRTECH
Mechanical/ Structural	Structure Weight WS 38.0	Manufacturing Complexity MCPLXS 5.8	New Structure NEWST 0.3	Design Repeat DESRPS 0	Equipment Classification MECID 0	Mechanical Reliability MREL .33
Electronics	Electronics Weight/ft^3 WECP	Manufacturing Complexity MCPLXE	New Electronics NEWEL	Design Repeat DESRPE	Equipment Classification CMPID	Electronic Reliability EREL
Development	Development Start DSTART 0190	1st Prototype Complete DFPRO C	Development Complete DLPRO C	Engineering Complexity ECMPLX 1	Tooling & Test Equip. DTLGTS	Prototype Activity PROSUP
Production	Production Start PSTART 0195	First Article Delivery PFAD C	Production Complete PEND C	Cost-Process Factor CPF 55	Tooling & Test Equip PTLGTS	Rate/Month Tooling RATOOL
Additional Data (Mode 10)	Electronic Volume Fraction USEVOL	Structural Weight/ft^3 WSCF	Target Cost TARCST			

Figure 35 Sample RCA PRICE Model Input Data Sheet

Price L Deployment File
Short Form

Deployment File Title ACTUATOR TRADE STUDY   Date: 6/10/81

Support Period (YR) 15

Number of Theaters 1

Short Form (0), or Long Form (1) 0

Data for Theater One

Number of Equipment Locations:	ED(1)	500						
Employment:	OTF(1)	.046000						
Number of Maintenance Locations:			OD(1)	18	DI(1)	6	DD(1)	1
Number of Supply Locations:	EDS(1)		ODS(1)	18	DIS(1)	6	DDS(1)	1

Data for Theater Two

Number of Equipment Locations:	ED(2)							
Employment:	OTF(2)							
Number of Maintenance Locations:			OD(2)		DI(2)		DD(2)	
Number of Supply Locations:	EDS(2)		ODS(2)		IDS(2)		DDS(2)	

Figure 36 RCA PRICE-L Model Deployment File

TABLE 32 RCA PRICE-L MODEL CALCULATED O & S VALUES

LRU MTBF, HOURS(MTBF)	1142.
LRU REPAIR TIME,HOURS(TF)	1.22
MODULE REPAIR TIME,HOURS(TM))	2.48
LRU PER SYSTEM,(EE)	18.
LRU COST,$(CUP)	1566.
MODULE COST,$(CMP)	671.07
PART COST,$(CPP)	21.65
PART COST ON--EQUIPMENT REPAIR,$(CPPE)	21.65
DEVELOPMENT COST,$(CEND)	2302638.
NON-RECURRING PRODUCTION COST,$(CPE)	1405651.
CONTRACTOR LRU REPAIR COST,$(CUR)	78.29
CONTRACTOR MODULE REPAIR COST,$(CMR)	234.88
MODULE TYPES,(P)	5.
PART TYPES,(PP)	95.
FRACTION NON-STD.PARTS,(FNSP)	0.50
LRU SUPPORT EQPT. COST,$(CFIM)	46254.
LRU+MODULE SUPPORT EQPT.,$(CFIP)	53784.
LRU S.E. FLOOR SPACE,SQ.FT.(FTSQF)	0.47
LRU+MODULE S.E. FLOOR SPACE,SQ.FT.(FTSQP)	0.55
LRU CHECKOUT TIME AT ORGANIZATION, HOURS(TC)	0.82
COST OF LRU CHECKOUT SET AT ORG.,$(CCOU)	18502.
LRU CHECKOUT SET FLOOR SPACE,SQ.FT.(FTSQC)	0.19

TABLE 33 RCA PRICE LCC SUMMARY - TYPICAL LRU

PROGRAM COST	DEVELOPMENT	PRODUCTION	SUPPORT
EQUIPMENT	$2303	$15496	***
SUPPORT EQUIPMENT	***	323	484
SUPPLY	***	310	1243
SUPPLY ADMIN.	***	5	80
MANPOWER	***	***	1994
CONTRACTOR SUPPORT	***	***	0
OTHER	0	***	16
TOTAL COST	$2303	$16134	$3817

TABLE 34 AIRPLANE ACTUATION TRADE STUDY LCC SUMMARY

	BASELINE		ALL-ELECTRIC		
QUANTITY	500	1000	500	1000	
RDT&E	15.8	15.8	13.9	13.9	
ACQUISITION	129.0	226.2	113.5	199.0	
OPERATIONS & SUPPORT	9.8	17.8	10.7	20.1	
TOTAL LCC	154.6	259.8	138.1	233.0	
LCC REDUCTIONS			16.5	26.8	11%

1. 1981 $
2. $ MILLIONS
3. 288 EH
4. OTF 1.4
5. 15 YEARS OF OPERATIONS

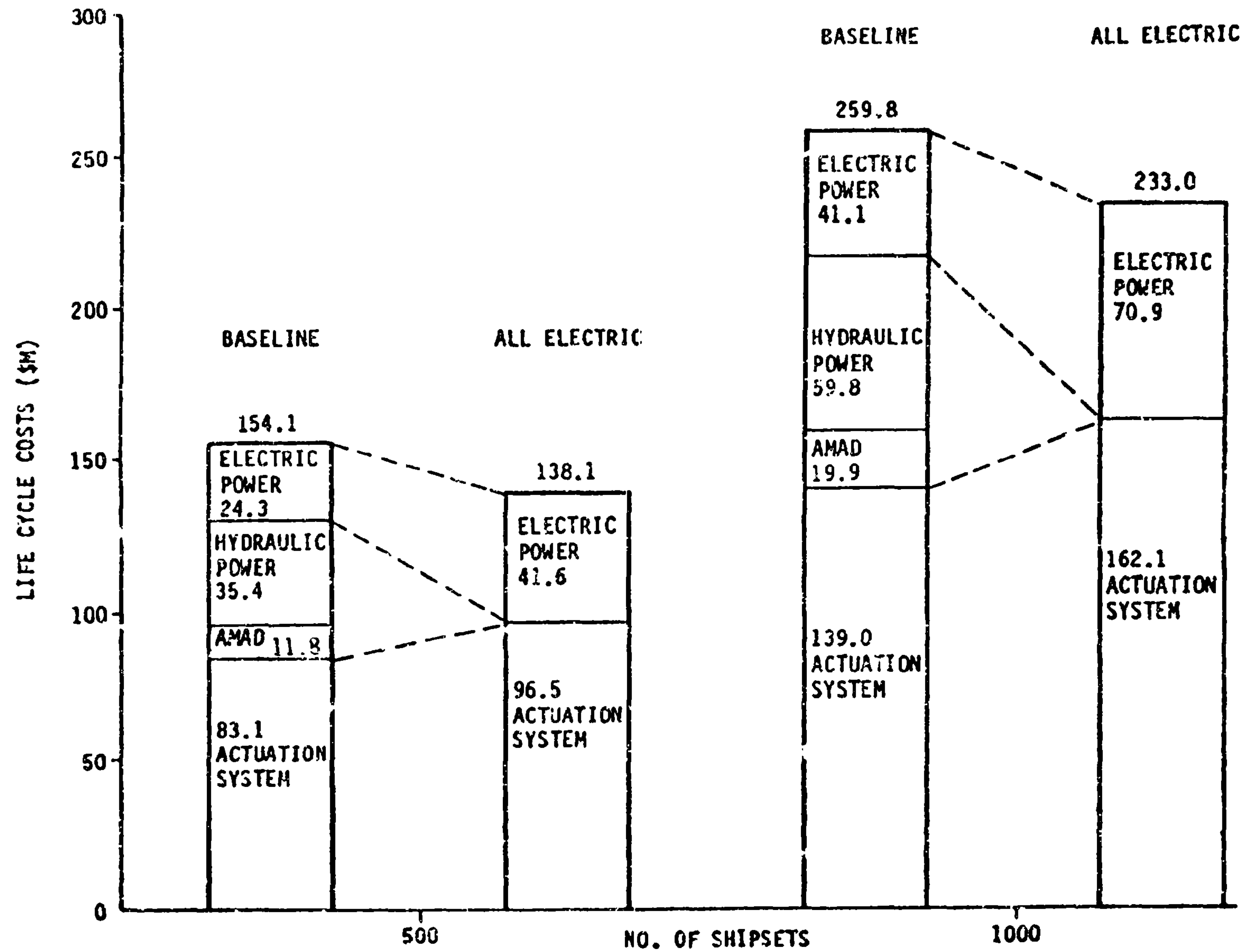


Figure 37 Airplane Actuation Trade Study LCC Summary

Table 35 and Figure 38 present the summary of actuation systems LCC for 500 and 1,000 aircraft and Tables 36 and 37 present detail actuation system LCC data for 500 and 1,000 systems respectively.

Table 38 and Figure 39 present the summary of secondary power systems LCC for 500 and 1,000 aircraft and Tables 39 and 40 present detail secondary power system LCC data for 500 and 1,000 systems respectively.

Although the total actuation system cost of the All-Electric Airplane is greater than the Baseline, the savings in the All-Electric secondary power system make it more cost effective overall.

5.4.8 LCC Sensitivity

A series of sensitivities were run to determine the sensitivity of the engineering judgments on the inputs for an LRU in the RCA PRICE Model. Runs were made for a range of Engineering Complexity, Manufacturing Complexity, New Structure, and New Electronics factors. Results are plotted as a percent change in cost.

Engineering Complexity is used in the RCA PRICE program to scope the development effort and to develop the amount of calendar time (in months) deemed necessary to complete the first prototype. For instance, a 1.0 signifies a new design within an established product line, continuation of the state of art, whereas a 1.6 signifies new design different from established product line, requiring in-house development of new electronic components or materials. The effects of these factors on RDT and E cost and total LCC are illustrated in Figure 40 and 41 respectively. Some changes in inputs affect all elements of cost, while others affect only one element. Engineering development complexity affects development cost as can be seen in Figure 40. However, manufacturing cost and operations and support cost remain approximately the same as can be seen in Figure 41. Other sensitivity runs were made on an individual LRU (the controller/inverter for the canard) with results as discussed in the following paragraphs. Figure 42 illustrates the effect on LCC of a plus and minus 50% change in weight of an LRU and Figure 43 shows the effect on LCC of a change in engineering complexity of an LRU.

TABLE 35 ACTUATION SYSTEM LCC SUMMARY

	BASELINE		ALL-ELECTRIC	
QUANTITY	500	1000	500	1000
RDT&E	8.6	8.6	9.8	9.8
ACQUISITION	69.9	122.1	79.4	138.5
OPERATIONS & SUPPORT	4.6	8.3	7.3	13.8
TOTAL LCC	83.1	139.0	96.5	162.1

1. 1981 $
2. $ MILLIONS
3. 288 EH
4. OTF 1.4
5. 15 YEARS OF OPERATIONS
6. INCLUDES INTEGRATION OF ACTUATION SYSTEM ONLY

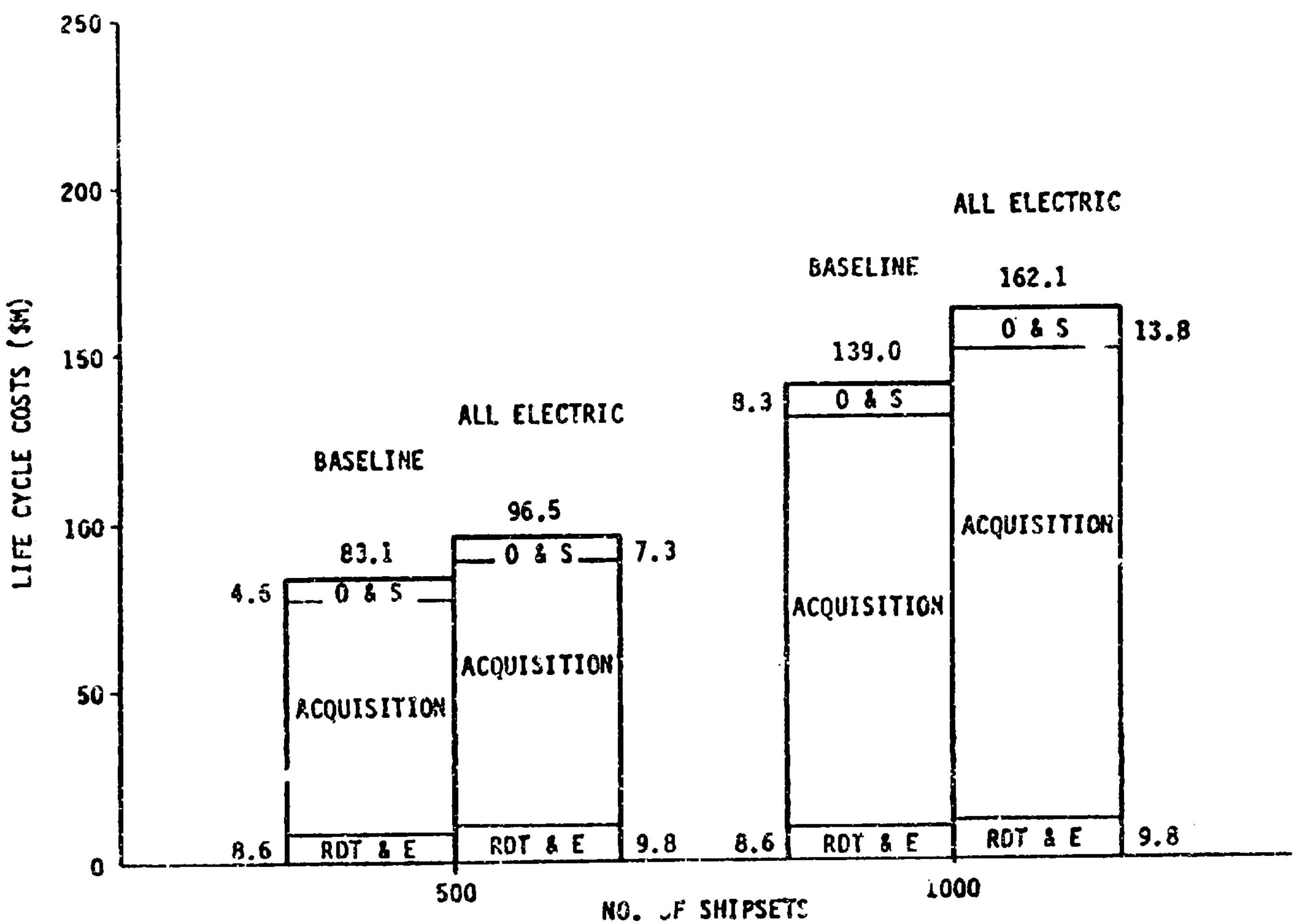


Figure 38 Actuation System LCC Summary

TABLE 36 ACTUATION SYSTEM LCC DATA (500 A/C)

	BASELINE				ALL-ELECTRIC			
	DEVELOP-MENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC	DEVELOP-MENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC
EQUIPMENT	8.6	67.7		76.3	9.8	76.3		86.1
SUPPORT EQUIPMENT		.3	.5	.8		.4	.6	1.0
SUPPLY		1.9	1.9	3.8		2.6	2.1	4.7
SUPPLY ADMIN.			.4	.4		.1	.8	.9
MANPOWER			1.4	1.4			3.4	3.4
CONTRACTOR SUPPORT			.4	.4			.4	.4
OTHER								
TOTAL COST	8.6	69.9	4.6	83.1	9.8	79.4	7.3	96.5

1. 1981 $
2. DOLLARS IN MILLION'S
3. 288 FH
4. OTF 1.4
5. 15 YEARS OF OPERATIONS
6. INCLUDES INTEGRATION OF ACTUATION SYSTEM ONLY

TABLE 37 ACTUATION SYSTEM LCC DATA (1000 A/C)

	BASELINE				ALL-ELECTRIC			
	DEVELOPMENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC	DEVELOPMENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC
EQUIPMENT	8.6	119.4		128.0	9.8	134.6		144.4
SUPPORT EQUIPMENT		.4	.7	1.1		.5	.7	1.2
SUPPLY		2.3	4.1	6.4		3.3	4.9	8.2
SUPPLY ADMIN.			.4	.4		.1	.8	.9
MANPOWER			2.7	2.7			6.8	6.8
CONTRACTOR SUPPORT			.4	.4			.5	.5
OTHER							.1	.1
TOTAL COST	8.6	122.1	8.3	139.0	9.8	138.5	13.8	162.1

1. 1981 $
2. DOLLARS IN MILLION'S
3. 288 FH
4. OTF 1.4
5. 15 YEARS OF OPERATIONS
6. INCLUDES INTEGRATION OF ACTUATION SYSTEM ONLY

TABLE 38 SECONDARY POWER SYSTEM LCC SUMMARY

	BASELINE		ALL-ELECTRIC	
QUANTITY	500	1000	500	1000
RDT&E	7.2	7.2	4.1	4.1
ACQUISITION	59.1	104.1	34.1	60.5
OPERATIONS & SUPPORT	5.2	9.5	3.4	6.3
TOTAL LCC	71.5	120.8	41.6	70.9

1. 1981 $
2. $ MILLIONS
3. 288 EH
4. OTF 1.4
5. 15 YEARS OF OPERATIONS
6. INCLUDES INTEGRATION OF SECONDARY POWER SYSTEM ONLY

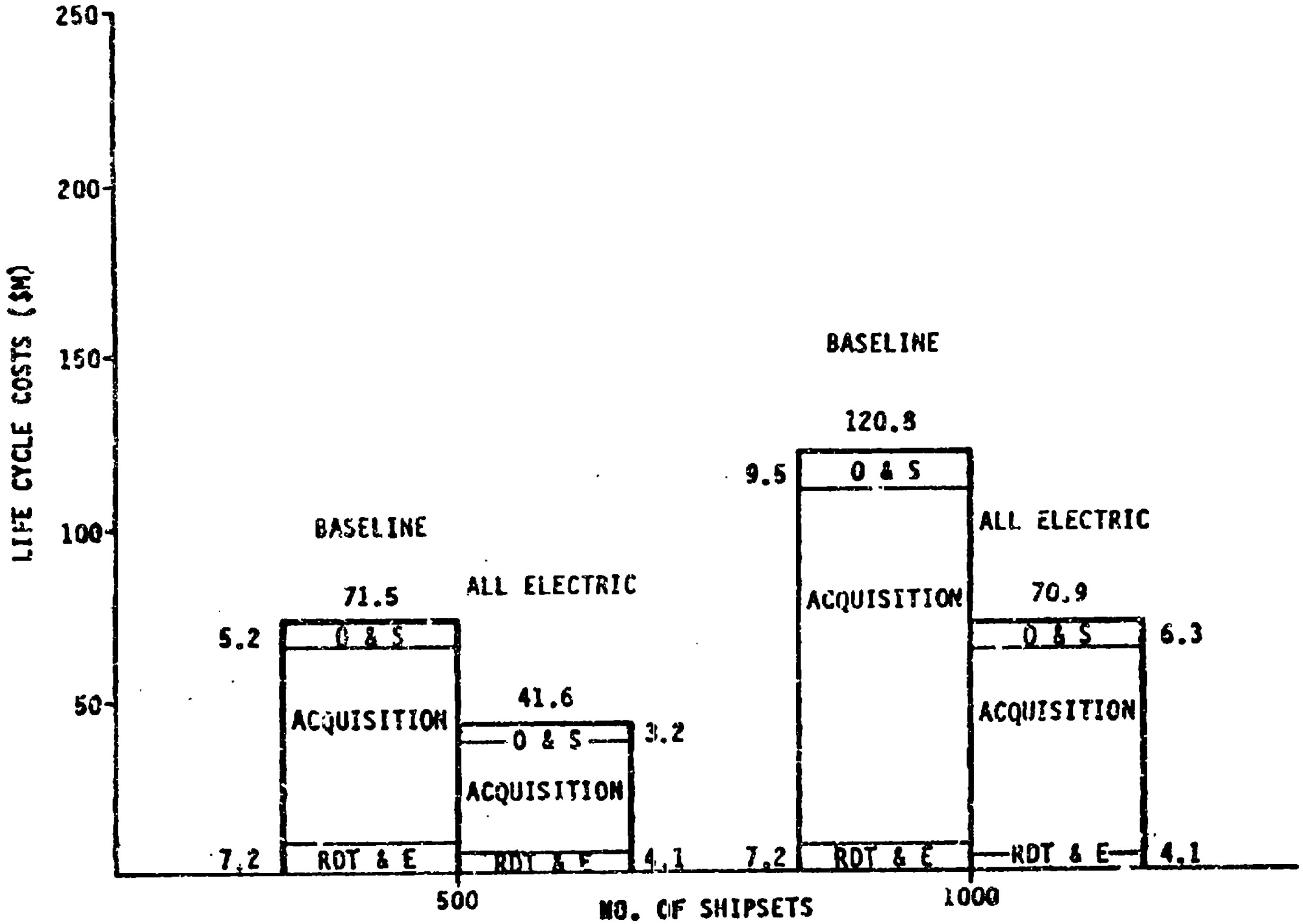


Figure 39 Secondary Power System LCC Summary

TABLE 39 SECONDARY POWER SYSTEM LCC DATA (500 A/C)

	BASELINE				ALL-ELECTRIC			
	DEVELOP-MENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC	DEVELOP-MENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC
EQUIPMENT	7.2	56.3		63.5	4.1	32.7		36.8
SUPPORT EQUIPMENT		.4	.5	.9		.2	.3	.5
SUPPLY		2.4	1.1	3.5		1.2	1.1	2.3
SUPPLY ADMIN.			.6	.6			.3	.3
MANPOWER			2.4	2.4			1.7	1.7
CONTRACTOR SUPPORT			.6	.6				
OTHER								
TOTAL COST	7.2	59.1	5.2	71.5	4.1	34.1	3.4	41.6

1. 1981 $
2. DOLLARS IN MILLION'S
3. 288 FH
4. OTF 1.4
5. 15 YEARS OF OPERATIONS
6. INCLUDES INTEGRATION OF SECONDARY POWER SYSTEM ONLY

TABLE 40 SECONDARY POWER SYSTEM LCC DATA (1000 A/C)

	BASELINE				ALL-ELECTRIC			
	DEVELOP-MENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC	DEVELOP-MENT	PRODUCTION	OPER. & SUPPORT	TOTAL LCC
EQUIPMENT	7.2	100.9		108.1	4.1	58.6		62.7
SUPPORT EQUIPMENT		.4	.5	.9		.2	.3	.5
SUPPLY		2.8	2.6	5.4		1.7	2.2	3.9
SUPPLY ADMIN.			.6	.6			.3	.3
MANPOWER			4.7	4.7			3.4	3.4
CONTRACTOR SUPPORT			1.0	1.0			.1	.1
OTHER			.1	.1				
TOTAL COST	7.2	104.1	9.5	120.8	4.1	60.5	6.3	70.9

1. 1981 $
2. DOLLARS IN MILLION'S
3. 288 FH
4. OTF 1.4
5. 15 YEARS OF OPERATIONS
6. INCLUDES INTEGRATION OF SECONDARY POWER SYSTEM ONLY

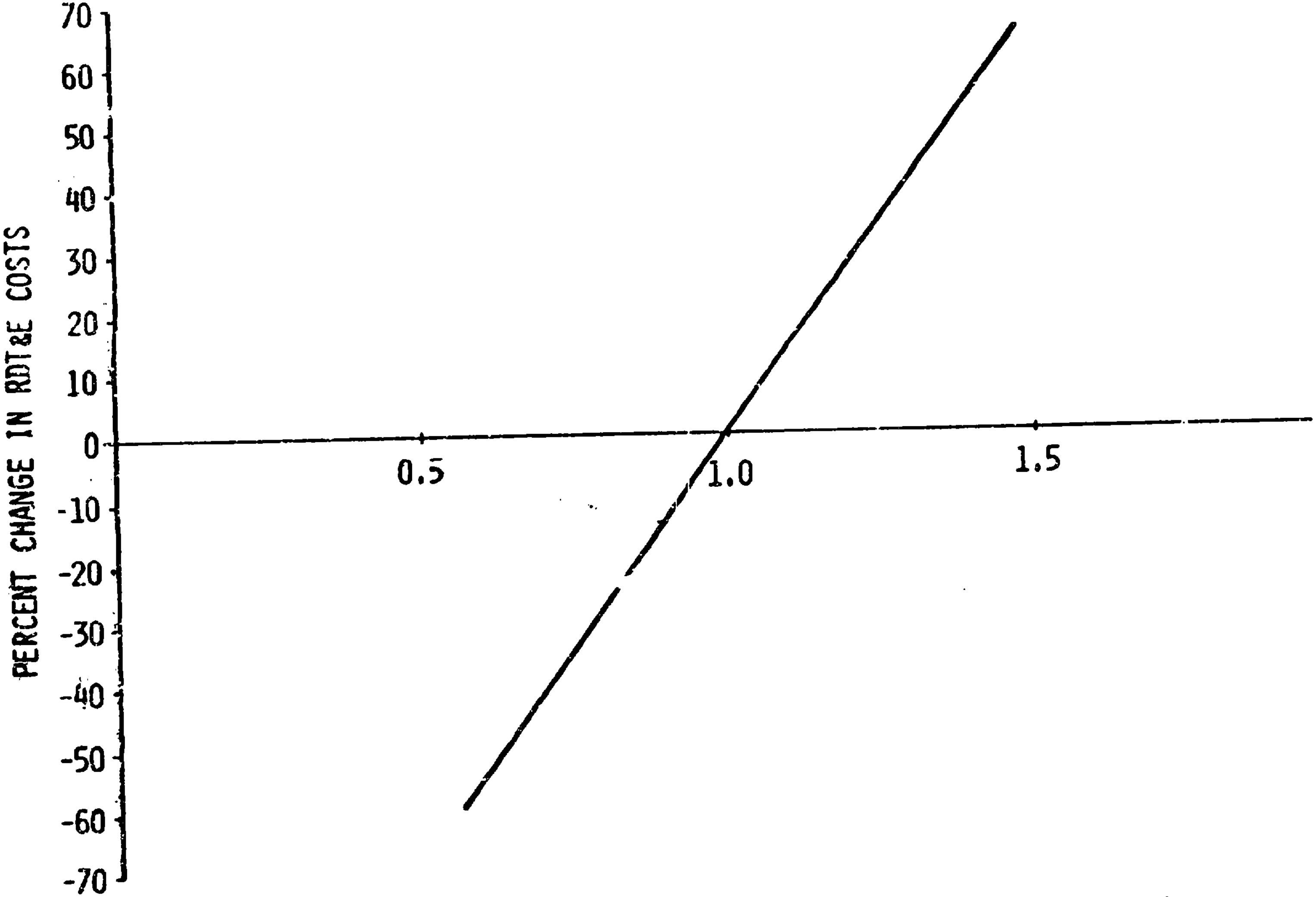


Figure 40 Sensitivity of Overall RDT&E Costs to Engineering Complexity

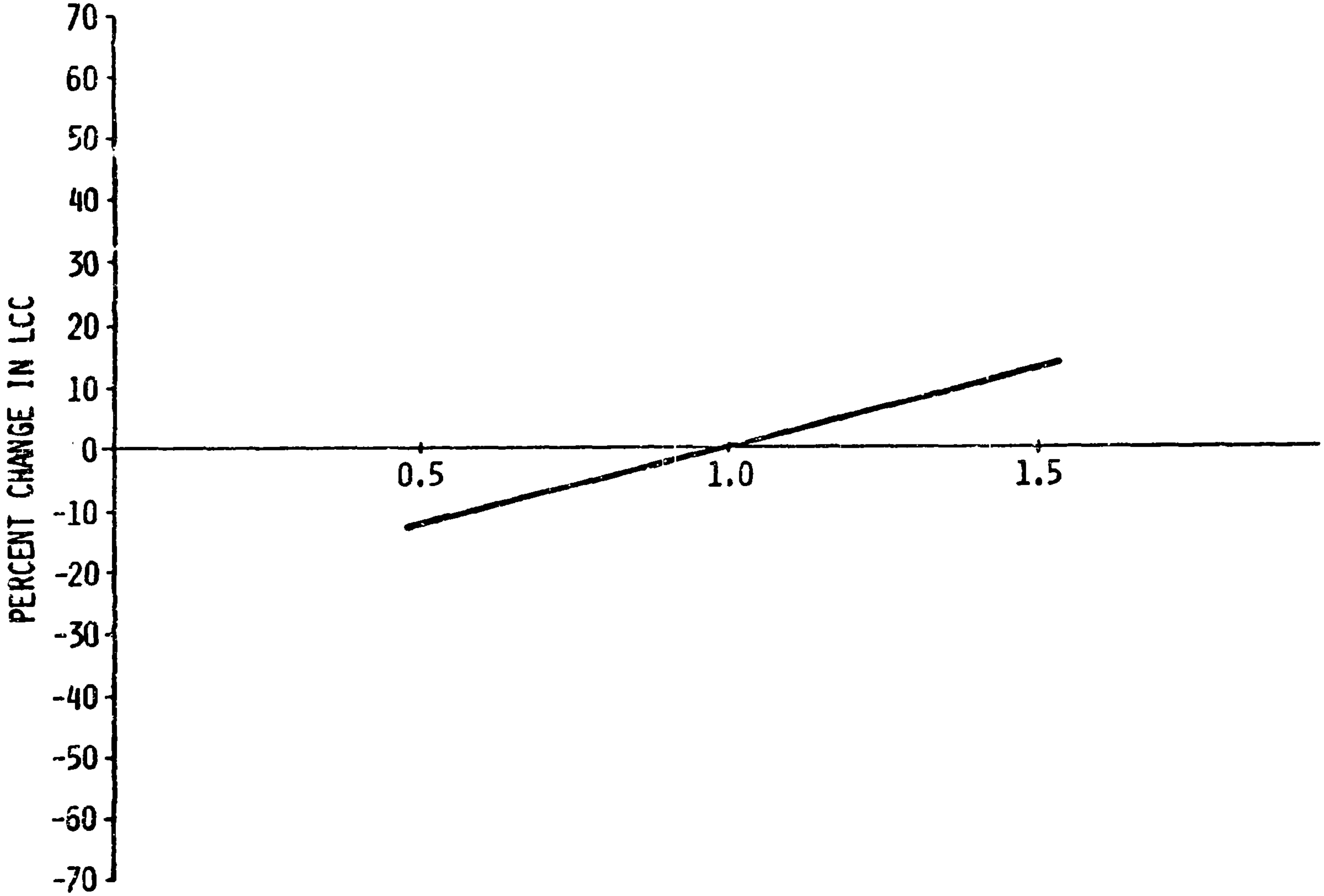


Figure 41 Sensitivity of Overall LCC to Engineering Complexity

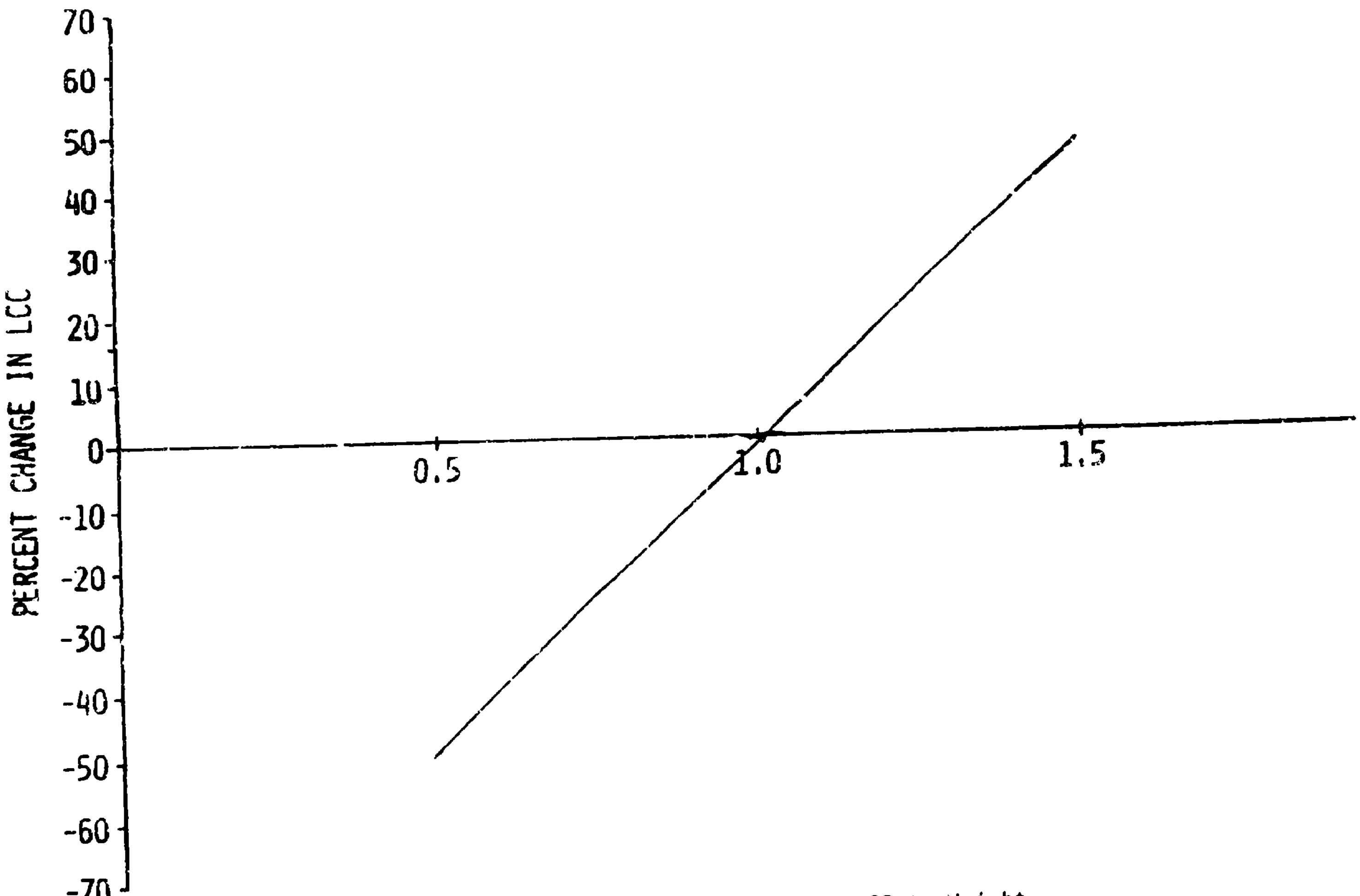


Figure 42 Sensitivity of LRU LCC to Weight

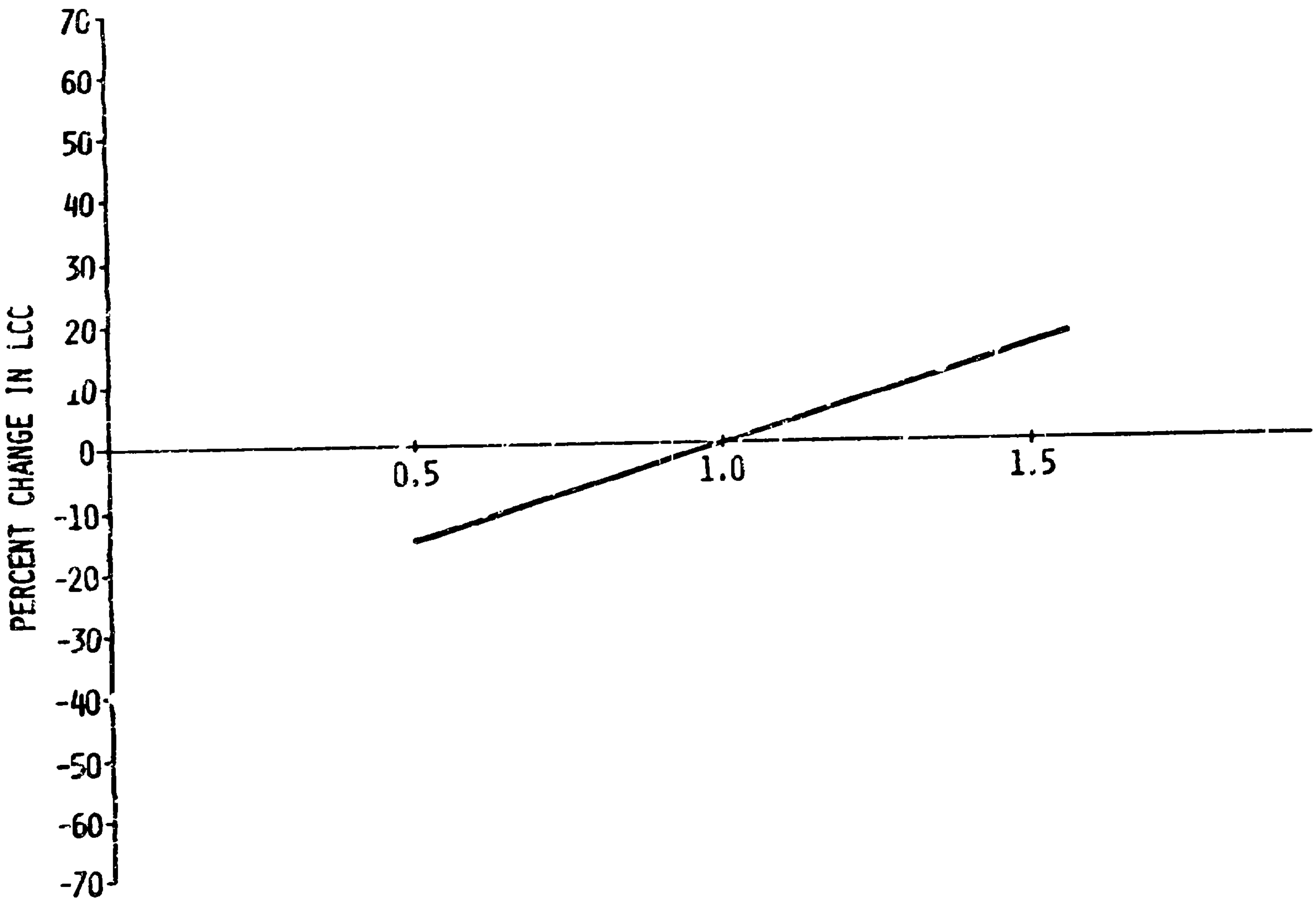


Figure 43 Sensitivity of LRU LCC to Engineering Complexity

Manufacturing Complexity of structure is usually an emperically derived value that represents the product's producibility. For instance, for an aluminum machined part a factor of 6.31 is used and for an aluminum forging a 5.77 is used. This factor defines the material, finished density, and fabrication methods. Manufacturing complexity of electronics is a complexity factor which is a function of its components, packaging density, manufacturing, testing, and power dissipation. For instance, a power supply composed of discrete components is assigned a factor of 6.941 and an LSI a factor of 7.368. Manufacturing Complexity Factors, as can be seen in Figure 44 and 45, affect LCC cost both for structural and electrical hardware. These figures show that cost growth for complexity is steeper for electronics than structural hardware.

New Structure and New Electronics defines the degree of new design required for the structure or electronics assembly that is unique. A factor of 0.1 indicates that 10% of the drawings are new. New Structure and New Electronics values only affect the development cost, as can be seen in Figure 46 and 47. The precentage of new structures and electronics does not affect the production and life cycle cost.

Electronic and structural next-higher-assembly-integration cost factors have no effect on LRU LCC cost.

[illegible] LCC sensitivities could have been run for the schedule factor but the [illegible] was assigned prior to the accomplishment of Phase I and the schedule [illegible] defined as 1990. The physical environment was also defined and no re[illegible]ty vs LCC sensitivities were performed.

5.5 Structural Integration

From a structural integration point of view the EM actuators may provide an advantage over the hydraulic actuators. This is because in most cases the most optimum method of actuation in hydraulic cases is usually the linear piston actuators. However, this may require out-of-contour fairings, or bell-crank mechanisms to couple to the surface being driven. In the case of EM actuation, hingeline actuators can be designed to fit within the wing surfaces. Also in the power extraction scheme utilized in the Baseline

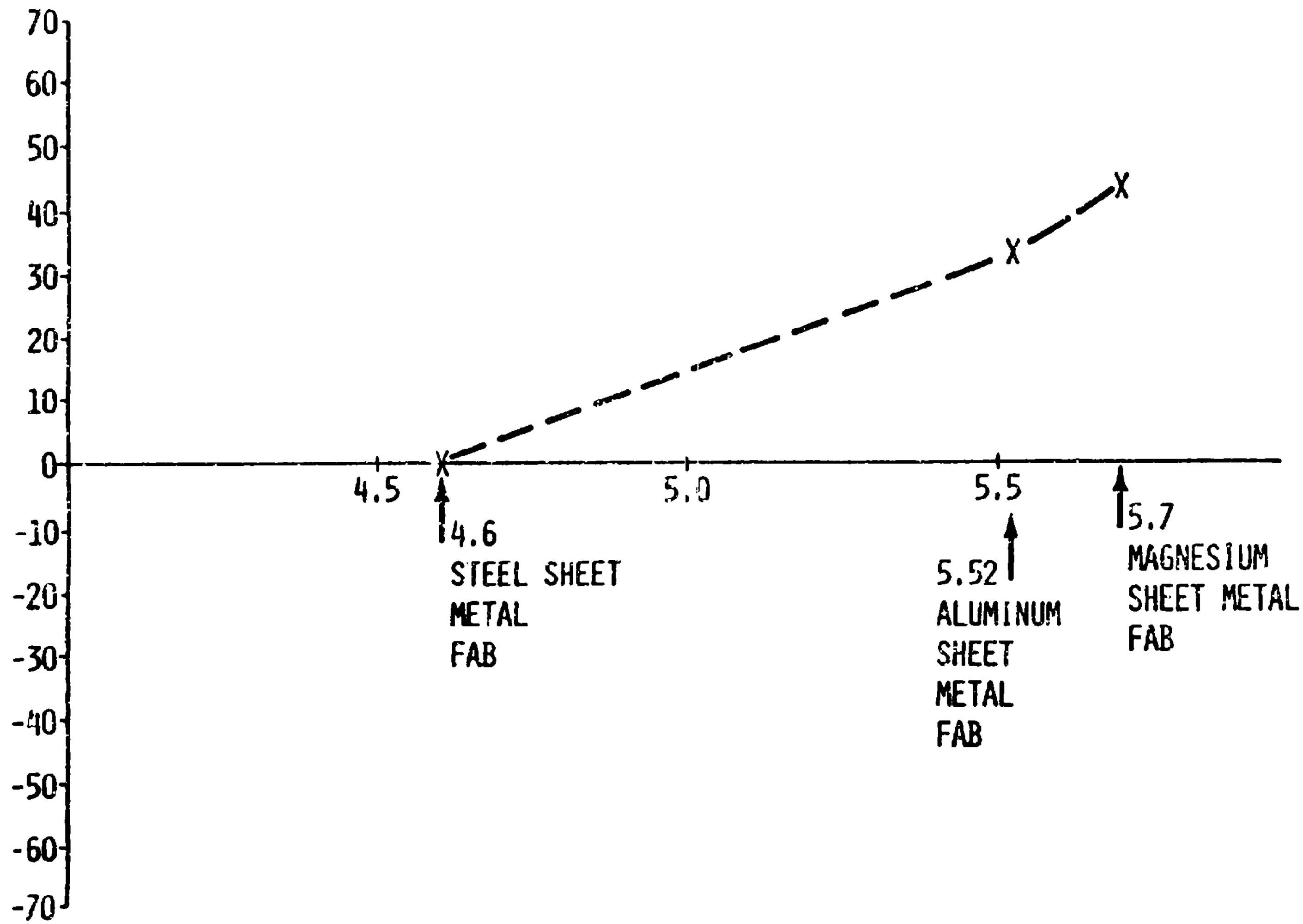


Figure 44 Effect of Material Change on LRU LCC

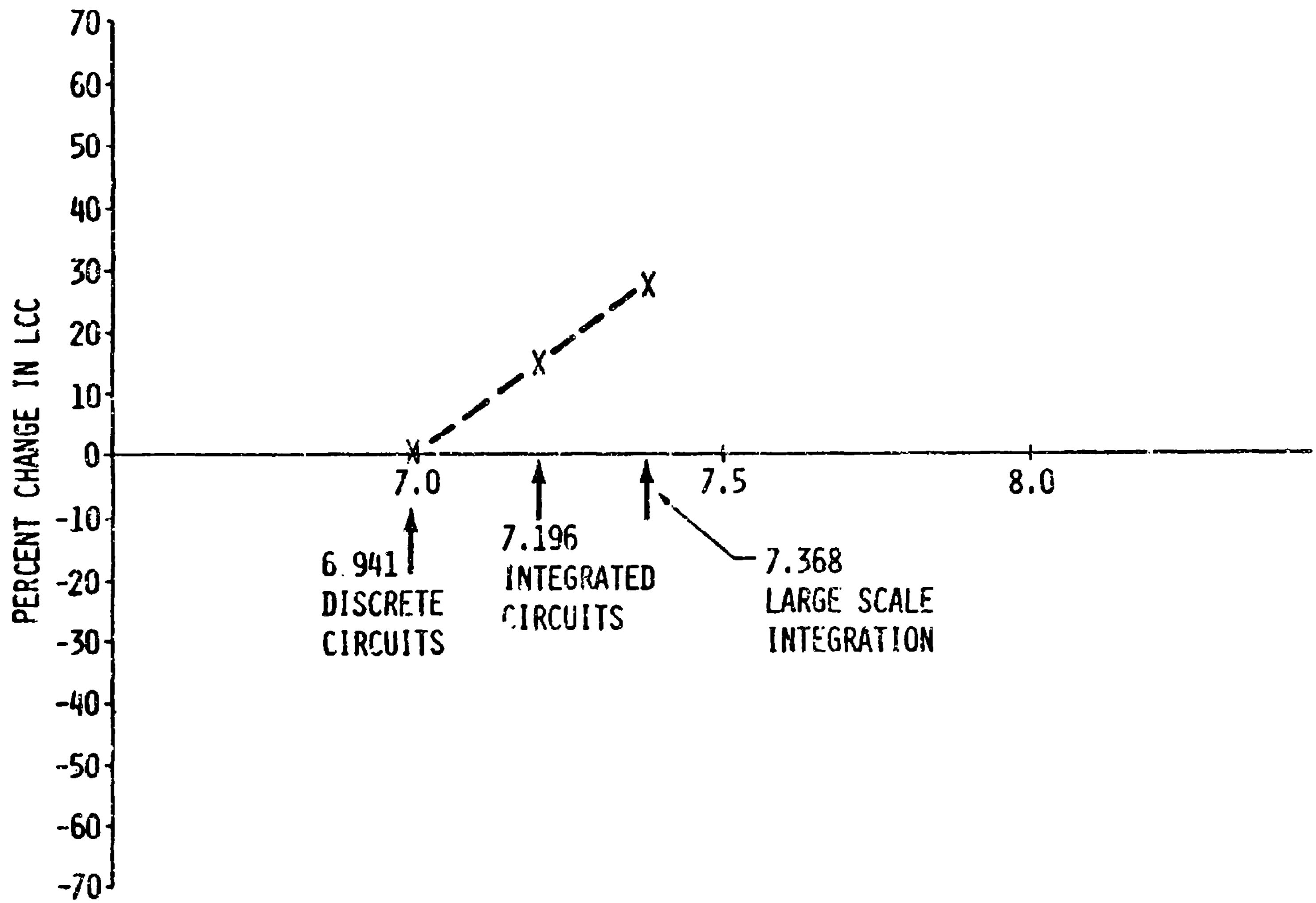


Figure 45 Effect of Electronic Circuit Design on LRU LCC

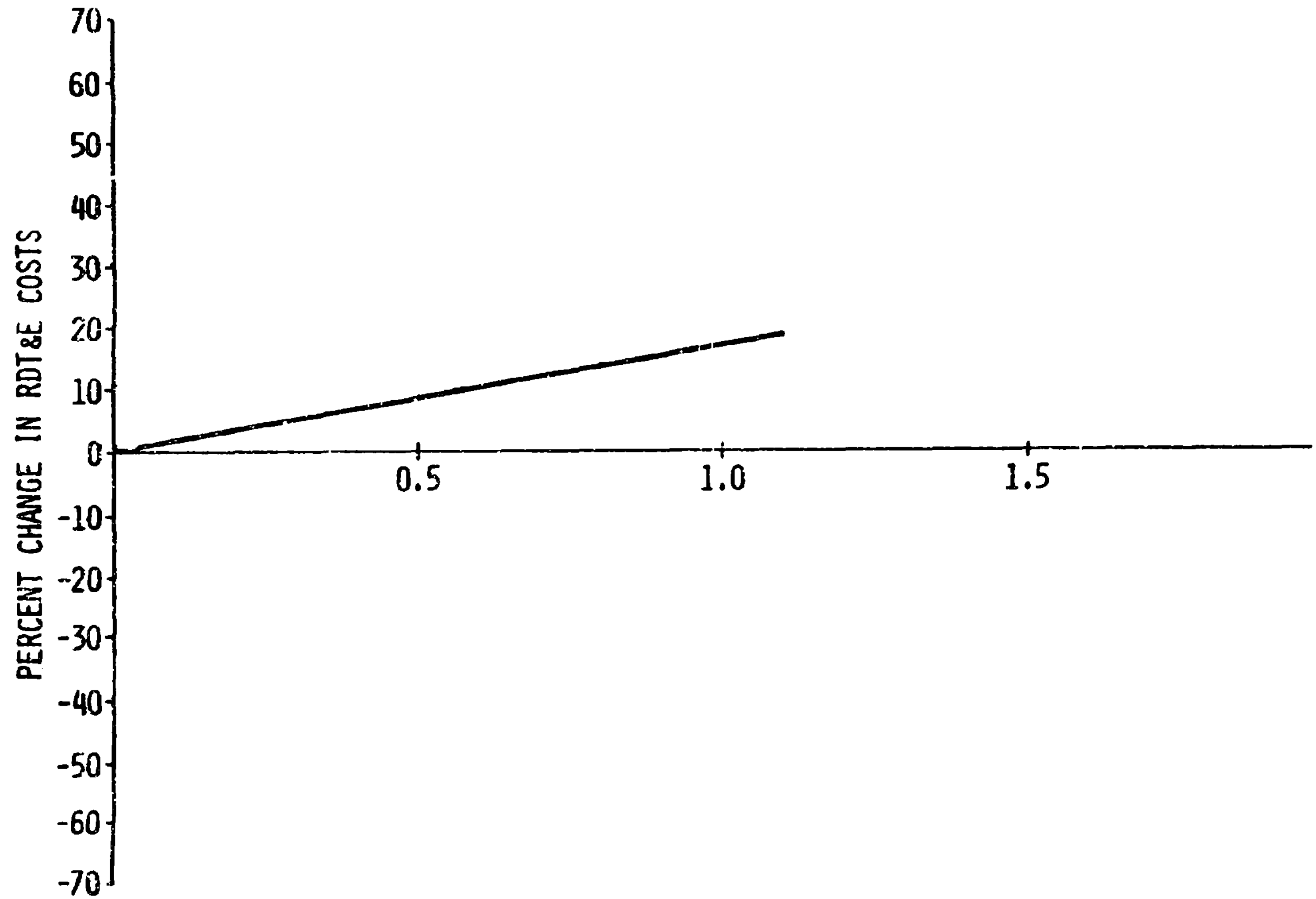


Figure 46 Sensitivity of LRU RDT&E Costs to New Structure Ratio

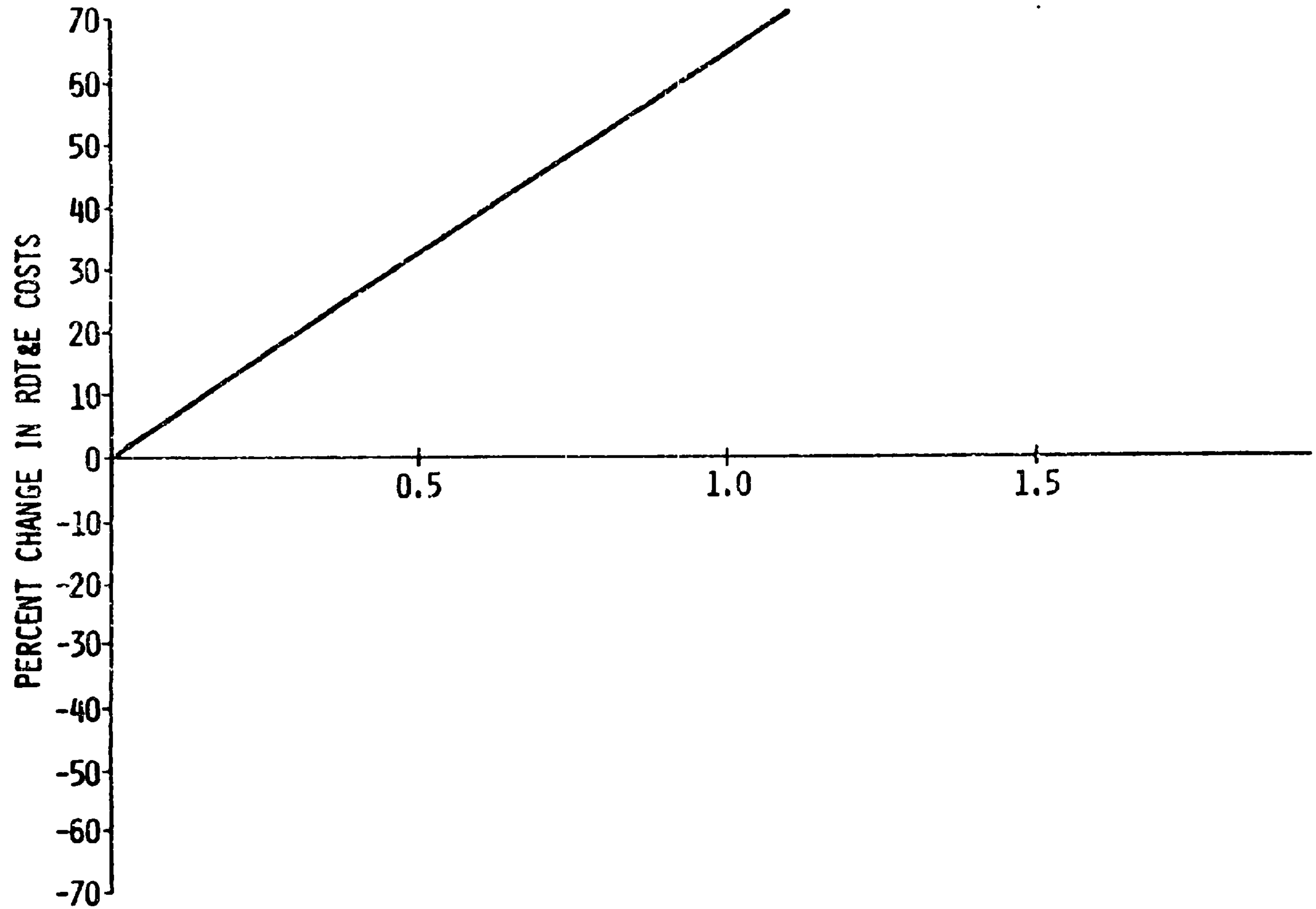


Figure 47 Sensitivity of LRU RDT&E Costs to New Electronics Ratio

Airplane via power take-off shafts driving AMADs, the location of the AMAD near the engines is desirable. This may be in an area where the fuel tank would need to be located for center of gravity adjustments during various phases of the flight. Also the structure has to be strengthened to support the AMAD hardware. In the case of the All-Electric Airplane, the power extraction is done electrically via a starter/generator which also performs the engine start function. The power conditioning equipment may be located where it will not interfere with the placement of the fuel tanks.

5.6 Growth

An evaluation of the growth capabilities of the actuation and secondary power systems of the Baseline and All-Electric Airplanes was conducted. The basic design philosophy utilized in the sizing of hydraulic actuators is to meet the specific requirements of the surface actuation. The piston cross section and stroke is sized for maximum load/stroke characteristics.

In the EM actuation systems the drive power is provided by Sm-Co permanent-magnet motors. The design of the motor is based on the average load. The motor can be driven to produce power levels above the capability of the average rating for short durations. The limiting factor is the heat generated under the various operating conditions. The motor windings should not be allowed to exceed a temperature which could damage the stator windings. This capability allows the EM actuation to satisfy additional peak load demands as long as the duration is compatible.

Hydraulic power systems are designed to meet military specifications such as MIL-H-5440. For example, the requirements for selection of engine-driven hydraulic pumps states that "a sufficient number of engine-driven pumps shall be provided to assure operation of control surface boost or power systems... ." Thus the sizing of the system is to assure that the basic requirements are met. If additional load growth occurs, the system would have to be resized.

On the other hand the sizing criteria for electrical power systems as specified in MIL-E-25499 states that "... the aircraft shall have a multigenerator primary electrical system which has a maximum continuous kVA capacity of at least twice the maximum continuous electrical load of the initial production aircraft." This sizing criteria allows for load growth capability in the electrical power system on both the Baseline and the All-Electric Airplanes. Due to the higher capacity generators on the All-Electric airplane, additional capability is available for short durations.

5.7 Survivability/Vulnerability

Survivability is assessed by examining the ability of the airplanes to safely withstand the following:

- Enemy action (combat survivability)
- All engines out
- Natural or induced environmental extremes
- Onboard system failures
- Maintenance errors
- Flight crew inaction or error

Although lightning is usually considered part of the "natural environment," this important subject is treated separately, along with electromagnetic compatibility, in paragraph 5.8.

The integrity of either aircraft is highly dependent on its powered actuation systems, especially those associated with the primary flight controls. A qualitative evaluation was made of the relative survivability of the Baseline Airplane versus the All-Electric Airplane with respect to their invulnerability to the factors listed above.

5.7.1 Combat Survivability

Techniques developed for enhancing electrical and hydraulic system invulnerability to enemy action fall into three main categories as follows:

(1) Design techniques for minimizing exposure so as to minimize the probability of a hit

- Avoidance of high susceptibility areas
- Use of shielded locations
- Concentration and protection of critical components
- Miniaturization of components
- Use of armor systems

(2) Damage resistant design techniques which minimize loss of function due to a hit

- Ballistic resistant materials and designs
- Fire/heat resistant materials

(3) Damage tolerant design techniques

- Redundancy
- Physical separation of redundant systems

Additional techniques that apply only to hydraulic systems are:

- Frangible actuators
- Actuator return - pressure relief devices
- Use of overboard drains
- Leakage protection devices such as hydraulic fuses and circuit breakers, isolation valves, reservoir level sensing and isolation circuits, and discriminating switching valves
- Reservoir considerations such as location, separation, and pressurization

Survivability of either airplane depends to a large degree on the invulnerability of critical systems to enemy action. In the Baseline this includes not only the actuation portion of the system, but also the hydraulic power supply to the system and the fly-by-wire electrical elements associated with the system. In the All-Electric Airplane, the vulnerability of critical systems is increased due to the added controller/inverters and associated cooling systems. On the other hand the wires supplying power to the actuation systems are slightly less vulnerable than the comparable hydraulic lines in the Baseline Airplane due to their smaller size. Overall, the All-Electric Airplane will require that emphasis be placed on the location and installation of the inverters and their cooling systems during the airplane design to insure the required level of survivability is achieved.

5.7.2 Non-Combat Survivability

The hydraulic systems on the Baseline Airplane of the 1990's should be comparable to hydraulic systems on current military aircraft relative to their high invulnerability to natural environments, onboard failures of other systems and equipment, maintenance error, and pilot and flight crew inaction and error. Methods of preventing failure of more than one hydraulic or electric power system due to other failures, including engine or tire bursts, and for preventing maintenance and other human errors are highly developed.

Invulnerability to induced environments should be somewhat better than current aircraft with engine-driven hydraulic pumps mounted directly on engine-mounted accessory gearboxes. The use of airframe-mounted accessory-drive (AMAD) gearboxes removes hydraulic pumps, valves, tubing, and hose from the high noise, vibration, and temperature environment of the engine compartment. The use of the higher (5,000-psi) system pressure should not introduce much of a problem for 1990-time period aircraft. There is a good backlog of successful operation of aircraft system operation at 4,000 psi, and many industrial systems operate at 5,000 psi. The Navy-sponsored testing at 8,000 psi has been quite successful; and, it is predicted that development of 4,000 and 5,000-psi systems for Air Force aircraft applications will be accelerated.

The electrical systems on both the Baseline and All-Electric Airplanes must be designed to be invulnerable to natural environments, onboard failures of other systems and equipment, maintenance error, and pilot and flight crew inaction and error because they supply the control and monitoring power to the fly-by-wire systems. In addition, the electrical system on the All-Electric Airplane must supply all actuation power to a level of redundancy comparable to the fly-by-wire requirement. However, the limits on power interruptions are not as stringent for actuation power as they are for fly-by-wire power.

The redundant actuators for primary controls on both airplanes are separated as much as possible. The hydraulic actuators on the Baseline Airplane are more jam tolerant while the EM actuators on the All-Electric Airplane are susceptible to a catastrophic jam in the gearing.

Either the Baseline or the All-Electric Airplane of the 1990's should be better able to maintain attitude control with all engines out than current aircraft. Upon loss of engine power, the LOX/JP-4 integrated power unit (IPU) can be brought up to speed in a matter of seconds (before the engines spool down) to supply the hydraulic and/or electrical power requirements. This is an important feature for an electric-command fully-powered-flight-control-system airplane, especially with the high-bypass-ratio engines of the future which are expected to have poor windmilling power capability.

In the 1990+ time frame, both the Baseline and the All-Electric Airplane will be fly-by-wire airplanes and will impose the same signal-level power requirements on the electrical power system in terms of redundancy and uninterruptible power. This is reflected in the two electrical power system schematic diagrams, Figures 16 and 28, where the Flight Critical Electronics (FCE) buses provide uninterruptible power while the three 28V DC buses and the battery provide the redundant power sources. All loads supplied by these buses are signal level or low power requirements.

The high power actuation loads are supplied by the triple hydraulic system in the Baseline Airplane and by the triple 270V DC bus system in the All-Electric Airplane. The third power source in the All-Electric Airplane is the flight-operable IPU generator which is started up whenever either main engine driven generator channel fails. Therefore, the loss of a single power source or any plausible single equipment failure will not result in permanent degradation of flight control system performance below FCS Operational State I, or temporary degradation below FCS Operational State II.

5.8 EMC/Lightning

The electrical power systems, digital systems, and electrical utilization subsystems for the two airplanes, and the electromechanical actuators for the All-Electric Airplane are designed to achieve EMC within the operating environments using the design guidelines of MIL-E-6051D, MIL-B-5087B, MIL-STD-461, and AFSC DH1-4.

5.8.1 EMC

A good equipment EMC design approach encompasses the whole compatibility problem from the circuit design concepts through the deliverable article. The objective is the marriage of complex circuits and equipment into a compatible system which operates within performance specifications in the specified environment.

Attention is given to the sources of noise generation within any equipment. This includes equipment designed for intentional radiation as well as that not specifically designed for radiation. Radio and radar transmitters may contain spurious oscillations, harmonics, oscillators, or products of these frequencies. Unintentional transmissions may result from broadband energy generation such as switching transients, commutation, rectifier and diode noise, and fast rise time waveforms. These unintentional transmitters can create very broad spectrums of high frequency components by a rapid change in voltage and/or energy level. A rapid change of one volt is easily sufficient to cause failure in meeting MIL-STD-461 EMI generation limits.

Equal attention is given the EMC environment. Circuits and equipment may be susceptible to interfering signals from the external electromagnetic field surrounding the installed equipment, signal input or output wiring, power supply wiring, or electromechanical systems.

In evaluating EMC for this trade study, the major variable element between the two airplanes is the addition of power-by-wire actuators and associated wiring on the All-Electric Airplane. Extra attention is given to these items since electromechanical actuators using solid-state switching for external commutation of the drive motors generate EMI noise that must be contained within the motor controllers to prevent conducted noise from interfering with operation of other power utilization equipment on the same bus, and to prevent radiation to nearby signal and control wires. In the electrical power generation systems the output rectifier/voltage regulator network of a permanent-magnet brushless DC generator and the cycloconverter in the VSCF system are both inherently EMI generators. However, since this has long been recognized, the designs of these devices include adequate shielding and filtering to contain the noise within the generator/converter assembly.

5.8.2 Lightning Protection

The interaction of lightning with an aircraft, either by direct strike or near-miss, induces electrical transients into the aircraft circuitry. Military aircraft of the 1990's will contain significant amounts of composite structure with poor electrical conductivity. In addition, the advanced electrical power and fly-by-wire systems used in these aircraft contain many solid state components. The combination of the two (reduced inherent shielding effectiveness of nonmetallic materials coupled with circuit components that have lower tolerance to electrical transients), presents design problems in both the Baseline and All-Electric Airplanes. The problem is intensified in the All-Electric case due simply to the added number of electrical circuits and wires.

Lightning induced transients present a hazard to electrical and electronic systems that is met by providing an adequate protection system. The occurrence of several direct lightning strikes plus many near-misses to a given aircraft during its service life is a certainty. A direct strike to an electrical circuit can result in considerable physical damage to the wiring as well as to circuit components attached to the wires. If the circuit is not struck directly, it will still have potentially damaging transient levels induced by magnetic coupling to the lightning currents flowing through the aircraft structure. These induced transients can have sufficient energy to damage or at least upset solid state components.

The mechanism whereby lightning currents induce voltages in aircraft electrical circuits is as follows. As lightning current flows through an aircraft, strong magnetic fields, which surround the conducting aircraft and change rapidly in accordance with the fast-changing lightning-stroke currents, are produced. Some of this magnetic flux may leak inside the aircraft through apertures such as windows, radomes, canopies, seams, and joints. Other fields may arise inside the aircraft when lightning current diffuses to the inside surfaces of skins. In either case these internal fields pass through aircraft electrical circuits and induce voltages in them proportional to the rate of change of the magnetic field. These magnetically induced voltages may appear between both wires of a two-wire circuit, or between either wire and the airframe. The former are referred to as line-to-line voltages and the latter as common-mode voltages.

In addition to these induced voltages, there may be resistive voltage drops along the airframe as lightning current flows through it. If any part of an aircraft circuit is connected anywhere to the airframe, these voltage drops may appear between circuit wires and the airframe. For metallic aircraft made of highly conductive aluminum, these voltages are seldom significant except when the lightning current must flow through resistive joints or hinges. However, the resistance of titanium is 10 times that of aluminum, so the resistive voltages in future aircraft employing these materials may be much higher.

Upset or damage of electrical equipment by these induced voltages is defined as an indirect effect. It is apparent that indirect effects must be considered along with direct effects in assessing the vulnerability of aircraft electrical and electronics systems. Most aircraft electrical systems are well protected against direct effects but not so well against indirect effects.

Until the advent of solid state electronics in aircraft, indirect effects from external environments, such as lightning and precipitation static, were not much of a problem and received relatively little attention. No airworthiness criteria are available for this environment. There is increasing evidence, however, of troublesome indirect effects. Incidents of upset or damage to avionic or electrical systems, for example, without evidence of any direct attachment of the lightning flash to an electrical component are showing up in lightning-strike reports.

While the indirect effects are not presently a major safety hazard, aircraft design and operations in the 1990+ time frame could increase the potential problem due to the following:

- o Increasing use of plastic or composite skin
- o Further miniaturization of solid state electronics
- o Greater dependence on electronics to perform flight-critical functions

Design of protective measures against indirect effects are being developed under USAF contract F33615-79-C-2006 (Reference 8).

5.8.3 Wire Routing for Lightning Protection

The primary reason for optimizing wire routing is to reduce the amount of electromagnetic flux coupled onto the conductors and therefore wiring is located as close as possible to the ground plane or structural frame. Exposed wiring (e.g., wires underneath a leading edge of a poorly conducting material) is routed close to the metal structure. The amount of flux that is coupled to a wire is proportional to the distance separating the two conducting mediums. Wiring is located away from apertures (e.g., windows) and regions where the radius of curvature of the airplane frame or outer skin is the smallest. In particular, wiring is not routed across obvious slots (e.g., access doors). Where full shielding is required, the cable is routed in an enclosed channel. Structural returns for exposed power wiring are avoided.

The primary threat to equipment is the conducted threat delivered to the equipment by:

a. Exposed interconnecting wiring, or
b. Interconnecting wiring attached to an exposed element (e.g., windshield heater circuit).

The only potential threat which depends upon the fields in the vicinity of the equipment is E-field coupling. I.e., nearby electric fields may induce a voltage upon the wiring terminating in a poorly-grounded case. In order of priority then, the rules for equipment placement are:

a. Equipment located to minimize exposure of interconnecting wiring.
b. Equipment located in areas which are shielded from electric fields induced by lightning; case well grounded to structure to minimize the E-field coupling.

5.8.4 Power Equipment Protection

At the present time, there are no power system requirements governing the suppression of lightning induced transients in the kilovolt range. If new specifications are imposed requiring the equipment to withstand the lightning induced transients presently observed, filtering or shielding of individual equipment would produce additional weight and cost problems in the overall airplane design. However, by increasing the transient suppression requirement in individual equipment from the present military specification of 600 volts to 6000 volts, the loss in electromagnetic protection from the usage of graphite composite materials would be less critical. A more viable solution is to either prevent the transient from being coupled on to the power feeders or to suppress the transient so it does not appear at the main power buses. Preventing the transient from appearing on the buses allows the use of equipment designed to the existing power quality standards. Methods to limit the lightning induced transients to levels below existing power quality standards are being developed (Reference 8). These methods include wire shielding, the use of TransZorbs™, varistors, zener diodes, filters, and surge arrestors, and the use of conductive coatings.

5.8.5 Airplane Comparison

Both airplanes are fly-by-wire and therefore require that particular attention be given to the electromagnetic compatibility and lightning protection of circuits and equipment associated with safety-of-flight. However, due to the additional electromechanical actuators and electronic controllers, considerably more design analysis and testing is required in the All-Electric Airplane to insure safety under all operating conditions and logical failure modes.

In evaluating EMC for this trade study, the major variable element between the two airplanes is the addition of power-by-wire actuators and associated wiring on the All-Electric Airplane. Extra attention must be given to these items since electromechanical actuators using solid-state switching for external commutation of the drive motors generate EMI noise that must be contained within the motor controller to prevent conducted noise from interfering with operation of other power utilization equipment on the same bus, and to prevent radiation to nearby signal and control wires. In the electrical power generation systems the output rectifier/voltage regulator network of a permanent magnet brushless DC generator and the cycloconverter in the VSCF system are both inherently EMI generators. However, since this has long been recognized, the designs of these devices include adequate shielding and filtering to contain the noise within the generator/converter assembly.

5.9 Environmental Constraints

Equipment on both the Baseline and the All-Electric Airplanes will have to be designed to withstand and operate satisfactorily in the following environmental conditions:

a. Temperature
b. Altitude
c. Humidity
d. Salt Spray
e. Sand and Dust
f. Fungus
g. Thermal Shock
h. Vibration
i. Mechanical Shock

Hydraulic Systems

Hydraulic systems and components have been designed to withstand and function under such environments for years. The one parameter which gives most concern is high temperature. High temperatures, due to supersonic flight or due to the use of hydraulic actuation of engine control functions such as variable-geometry inlets and exit nozzles, may require special fluids and seal materials which will not break down due to sustained thermal exposure. Many supersonic aircraft, such as the F-111, F-14, F-15, F-16, F-18, and B-1, use standard petroleum fluid per MIL-H-5606 and standard Buna-N nitrile O-ring seals. Other aircraft, such as the B-58, B-70, and Concorde SST, were designed for use with silicate ester fluids and either special neoprene elastomer seals or all-metal seals (B-70). The Mach-3 SR-71 uses a synthetic hydrocarbon with all-metal seals; and, the X-20A (Dyna Soar) controlled-reentry manned orbital space vehicle was also designed with that fluid and with a combination of metal seals and high-temperature elastomeric seals. The engine-control hydraulic system on the B-70 was designed with a chlorinated silicone fluid and operated at some 600°F fluid temperature.

One distinct advantage of distributed hydraulic systems is that they are easily cooled. The fluid return lines can be circulated through fuel tanks to conduct their heat load to the lower temperature fuel, or through fuel-to-oil heat exchangers to take advantage of the higher thermal film coefficients caused by the flow of fuel to the engines.

Electrical Systems

Electrical power generation and distribution systems have been designed to withstand and operate in aircraft environments such as listed above. Electronic equipment items have to be provided with adequate cooling to maintain internal temperatures at which the reliability of the semiconductors are not impacted. Certain precautions are also necessary to locate equipment in areas where it will not be exposed to extremes of the above listed environments. During the design of the aircraft, adequate consideration has to be given to location of sensitive electronic equipment in areas where ambient conditions will subject the equipment to a minimum of environmental extremes.

In an All-Electric aircraft, the EM actuators will be located in areas which will be at one or more of the environmental extremes listed above. An example of this is the location of EM actuators in the leading and trailing edge surfaces of the wings. Here the actuators are subjected to the temperature extremes (especially high temperatures at supersonic cruise conditions). The worst case temperature in the leading edge is 275°F at the upper surface. The EM actuators must be designed to withstand and operate at these temperatures. Temperatures in excess of these values may require that additional cooling be supplied. Other environments such as salt spray, sand and dust, vibration and shock extremes will also impact the design of the EM actuators. Although these environments will impact the design of the EM actuators, none of them are too severe to preclude the use of EM actuation.

5.10 Technology Risk

The Baseline Airplane secondary power generation system is similar to that proposed for the Boeing supersonic transport and later incorporated in the B-1 and F-15 aircraft. The airframe-mounted accessory-drive (AMAD) gearboxes are well-proven designs which provide a great deal of operational capability. They allow hydraulic and electric system checkout and operation on the ground without operation of the main engines. The integrated power unit can drive all of the hydraulic pumps and generators.

The LOX/JP-4 integrated power unit (IPU) allows fast engine starts both on the ground and in flight at any altitude. It is currently under development by the AFWAL Aero Propulsion Laboratory, Aerospace Power Division, Power System Branch. It combines the performance of a bipropellant turbine power unit with a conventional gas turbine APU and should be sufficiently developed for aircraft use by 1985.

The hydraulic system pumps and other components are all based upon proven technology. Several aircraft hydraulic systems have been put into production with 4,000-psi operating pressures, and many industrial equipments use 5,000-psi systems. The use of 15V-3Cr-3Sn-3Al titanium alloy for hydraulic tubing has yet to be proven. It has an ultimate tensile strength of 200,000 psi compared to 125,000 psi for the 3Al-2.5V cold-worked titanium alloy currently in use, and offers a 37.5% reduction in dry weight in the larger sizes. In the smaller sizes, 3/16 and 1/4-inch diameter, the same wall gage (0.016 inch) as would be used with the 3Al-2.5V tubing was assumed. Although the 15V-3Cr-3Sn-3Al alloy has yet to be applied to hydraulic tubing, its physical properties appear compatible to that application; and, it is considered to be the material of the future.

The hydraulic actuation systems are all based upon proven actuator, hydraulic motor, and electrohydraulic servovalve designs. The only unique features are the use of valves to sequence the canard ram actuators and elevon ram actuators in stages depending upon the imposed aerodynamic hinge-moment load, and the use of digitally-controlled externally-commutated hydraulic motors operating through a torque-summing gearbox for the rudder. These are two types of load-adaptive actuation system arrangements being investigated by the Boeing Military Airplane Company.

Electromechanical actuator designs for the All-Electric Airplane include light-weight low-torque high-speed electric motors along with high-ratio speed reducing gearboxes and ballscrews. The electric motors require high energy produc. Sm-Co permanent magnets. The availability of magnets with large energy products (22 to 30 megagaussoersted)at reduced cost and increased volume will be necessary. Increasing motor speeds will result in reduced motor size and weight for a fixed power requirement. Motors used in this study were in the range of 18,000 to 25,000 rpm. While motor speed is not limited by existing technology (units in excess of 100,000 rpm have been built), there is certain risk associated with the motor and gear train technology, especially when the actuator is to be utilized for random duty cycle applications such as for primary flight controls. The gearboxes can be jammed due to loss of lubricant, gear wear, bearing wear, galling failure, fretting corrosion, or tooth breakage. Improvements are needed in gearbox design and overall acutation efficiency.

Electromechanical actuators of this type are being used on the Air Force/Boeing AGM 86A (Air Launched Cruise Missile) for the fin control. Electro-mechanical actuators were also used on the Compass Cope remotely piloted aircraft. However, these were low horsepower units.

The equipment used for electrical power generation in both the Baseline and All-Electric Airplane is based on recently developed technology. The 60 and 150 kVA permanent magnet starter/generators have been built or are in the development stage under programs being conducted by the AFWAL Aero Propulsion Laboratory. A flight test of a 60 kVA starter/generator in conjunction with a Variable Speed Constant Frequency (VSCF) system is planned for the near future. The Baseline Airplane power conditioning and distribution system consists of a 115V AC 400 Hz VSCF system. This type of system has already flown on certain versions of the A-4 and also the F-18 aircraft.

The All-Electric Airplane power conditioning and distribution is done at 270V DC. This type of equipment is also under development under funding of the Naval Air Development Center. The major risk involved in this area is in control, protection and switching of large currents at 270V DC and in the integrity of the redundant power bus. Development in this area is also being conducted and some protection and switching hardware has been built and demonstrated.

VI TECHNOLOGY NEEDS

This trade study assumed a state-of-the-art existent in the 1990 time frame, and therefore concepts envisioned to be available in the 1990 time period were exploited in the study. Consequently, there are inherent technical needs involved in the results of the study, based on the fact that a mature technology based was assumed.

Because of the years of experience and solid technology base that exists with hydraulic controls and actuation systems, and the lack of equivalent experience, and therefore relatively weak technology base with electric controls and actuation, there are greater technical needs associated with the All-Electric Airplane. This does not mean that nothing needs to be advanced in the Baseline Airplane, but only that there are less risks involved in achieving the Baseline Airplane relative to the All-Electric Airplane.

The technology needs to achieve both airplanes are discussed in the following paragraphs.

6.1 Baseline Airplane Technology Needs

6.1.1 Actuation Systems

The use of load adaptive/stored energy actuation systems could significantly reduce equipment weight and so the development of these systems should be pursued.

Multiple-piston motors can be used in some applications with little or no gearing and could account for additional weight savings.

The development of a staged sequential actuation system would be desirable. In this concept multiple hydraulic ram actuators are sequentially controlled in a way which allows them to adapt their power demands to meet the magnitude of resisting loads and also to recover power from aiding loads. The advantage is that the demand from the supply pump is directly reduced by the number of actuators in the group.

The use of high pressure hydraulic systems contributes to a reduction in hydraulic system weight. The developments required in this area are high pressure pumps, seals, tubing, and fittings.

6.1.2 Special Hydraulic Component

The flexibility and reliability of a hydraulic power system can be improved by the use of a high-flow bidirectional power transfer unit. This unit, connected between two hydraulic power systems, can provide a second source of power for each of the systems and therefore is a desirable technology advancement.

The development of hydraulic fuses and circuit breakers will improve airplane survivability by providing means to isolate failed hydraulic systems and limit fluid loss after sustaining physical damage.

Direct-driven single-stage servovalves are currently under development and have the potential for reducing the internal fluid leakage and power loss associated with two-stage valves. Additional development is needed, however, to provide the driving force capability to overcome jamming due to contaminants in the hydraulic fluid.

The use of digitally-controlled stepper-motor-driven rotary distribution valves with hydraulic-motor-driven actuation systems and the use of staged sequentially-controlled valves with multiple cylinder piston actuators have the potential for significantly reducing peak hydraulic system flow demands. The potential gains warrant further development.

6.2 All-Electric Airplane Technology Needs

6.2.1 Motors

The availability of magnets with large energy products at reduced cost and increased volume will be necessary for future servo systems. An energy product of 22 x 10^6 gauss-oersted was used during the study. Energy product magnets above the study value (30 x 10^6 gauss-oersted) with improved properties would be welcomed. The availability of such magnets in commercial quantities will allow the development of smaller, lighter motors, with higher specific power and power-rate capabilities.

Increasing motor speed is desirable in that it reduces motor size and weight for a fixed power requirement. For study purposes, an upper limit on motor speed of 25 Krpm was used. While motor speed is not limited by existing technology (units running in excess of 100 Krpm have been built), questions concerning motor and gear train reliability remain to be answered. This concern is especially valid for random duty cycle machinery such as position servos.

Numerous parameters must be specified during the motor design process. Attempting to satisfy all of the actuation system requirements with an optimum motor design is an exceedingly difficult engineering task. Frequently, motors are overdesigned because of this; occasionally a motor is underdesigned resulting in inadequate performance or failure. Development of motor selection criteria or algorithms for servo applications would be very beneficial to the designer. Such tools would allow rapid preliminary design, and expanded detail design capabilities for motor optimization.

Maintaining the largest possible rotor l/d ratio is desirable, in that it minimizes rotor inertia, thus maximizing motor acceleration and power-rate. A maximum l/d of 3:1 was used as a design constraint during the study. Building motors with such large l/d ratios, while feasible, is difficult. Improved manufacturing methods permitting exploitation of favorable geometries is viewed as being desirable.

6.2.2 Electronics

Power FETs with the required characteristics must be developed in order to satisfy control and thermal management schemes. A suggested device rating of 50 amps is conservative, and should be readily achievable during the next decade.

Although judicious design of a power controller/inverter can avoid damage due to switching transients, the problem of inductive energy dissipation must be dealt with. Bus-to-controller and controller-to-actuator line inductance will determine energy dissipation requirements (snubber circuit design) and motor response characteristics (electrical time constant). Both of these inductance sources will be driven by bus characteristics, and controller-actuator location.

Additionally, over-voltage conditions due to motor over-speed (e.g., response with aiding load) must be addressed. Again, controller/inverter design will provide a path for power flow and energy dissipation, but bus characteristics will be a major factor in determining configuration.

Compact, reliable optical/electrical interfaces are currently available. However, application of these interfaces in FCS equipment has yet to be demonstrated. The application of optical/electrical interfaces at the FCS actuation system controllers, inverters, and actuators; and optical data transmission between these assemblies must be evaluated and demonstrated.

Present microprocessors are adequate for the proposed application. Increased through-put capability and environmental operating conditions would be desirable, from the standpoint of application and reliability.

6.2.3 Controller/Inverter Thermal Management

Further work remains to be done in the areas of controller/inverter optimization and analysis.

Long term usage of R-113, -11, or some other coolant must be addressed. Resistance to chemical decomposition, and maintenance of a high dielectric rating are necessary for application to controller/inverter cooling.

A careful evaluation of the heat sink employed (air, fuel, or other) must be performed for each application. The selection will impact the aircraft in both weight and power demand.

Methods to reduce the internal thermal resistance of the semiconductor devices should be investigated. The internal thermal resistance contributes a significant portion of the overall resistance between the junction and cooling medium.

6.2.4 Mechanical Components

Operating stresses of approximately 90 and 140 ksi were used for the gearheads and hingeline drives respectively. These stress levels are at or slightly ahead of the state of the art. The smaller hingeline drive used for the elevon, spoiler, and rudder would operate at a maximum stress level of 179 ksi. The drive would have a life of approximately 10,000 cycles at the corresponding load (fully reversed cycling, 90 percent confidence factor).

Advances in material fatigue characteristics will be required, if the life or confidence factor for designs such as the above are not adequate for a given application.

The impact of increased gearing speeds should be investigated. For the speeds assumed during the study, oil sling lubrication would be necessary. This could impose sealing and maintainability difficulties.

Measurement of drive stiffness, static and dynamic, is very difficult due to the stiffness values, loads, and frequencies involved. Development of test methods with repeatable (to within some scatter factor) results, would lessen the almost total dependence on calculated data.

6.2.5 Control

Improved sensors for motor rotor position and rate, and actuator position and rate are necessary. Current devices have characteristics which lead to vagueness during a change of state (step outputs) or nonlinearity (proportional outputs). Sensors which provide a direct digital input would be the most desirable, since A/D converters would be unnecessary. Optical sensors would allow direct coupling to a controller bus.

In the event of a failure in one channel of a multi-motor actuator, control reconfiguration will be required. This requirement may be likened to a "multi-mode" adaptive control. Development of adaptive control schemes to deal with actuation system failures will be necessary. Implementation of adaptive control would also allow its expansion to full time adaptive control for selected parameters.

Modern control theory has matured during the past two decades into a useable control methodology. A considerable body of literature has developed, as a result (Reference 6). However, due to unfamiliarity or computational difficulty, most servo engineers have preferred to utilize classical control theory for design purposes. The literature of modern control theory should be reviewed for applications to servo design. A partial motivation for this recommendation is that EM actuation control systems are inherently nonlinear; and many of the components have nonlinear characteristics which dominate the response. Modern control theory is much better equipped to deal with nonlinear control systems than is classical theory.

The design of digital (discrete) controls is no more complex than analog (continuous) controls; and as common place as analog controls of ten years ago. While the technology has advanced, relevant specifications have not changed (Reference 7). A desirable advancement would be to update applicable servo references and specifications to address both digital and analog control schemes.

6.2.6 Secondary Power System

In the area of secondary power systems, studies will have to be conducted to determine the best method of providing electrical power to EM actuation systems. This will include effort in the following areas:

- Studies to determine the type of power to be generated and distributed and the level of power conditioning needed.

- Type of generation system that would be most amenable to perform the engine start function.

- The best means of extracting power - whether the generator should be mounted on the engine spinner vs on a gearbox connected to a power takeoff shaft.

VII RESULTS AND CONCLUSIONS

7.1 Discussion of Results

The objective of the design effort was to ensure that the actuation and secondary power systems for both airplanes meet all the design requirements.

In the first phase of this program, actuation systems requirements for the various functions were defined. During the second phase of this contract, actuation systems were configured for the various applications to meet the requirements specified in the first phase. Also during the second phase, secondary power systems were configured to power the actuation functions, in addition to meeting all the other airplane secondary power requirements. From these configurations an optimum set of actuation and secondary power systems was selected for both the Baseline and All-Electric Airplanes. Boeing's experience in the design and use of hydraulic actuation systems, along with that of leading industry suppliers, provided the basis for final configuration selection for the Baseline Airplane. In the case of the EM actuation systems, the final selections were made based on information supplied by the subcontractor, AiResearch Manufacturing Company of California. AiResearch also performed analyses of the flight control EM actuation systems to make sure that these systems met all the requirements specified in the first phase of this program. Thus, there was a good level of assurance that the two sets of systems that were traded in the third phase would meet all the performance requirements.

A summary of the quantitative comparisons of the Baseline and All-Electric Airplane systems is shown in Table 41. The weight of the EM actuation systems was about 20% higher than the weight of the hydraulic actuation systems. On the other hand the weight of the secondary power system of the All-Electric Airplane was 20% lower than that of the Baseline Airplane. Overall the total weight of the actuation and secondary power systems was about the same for the two airplanes.

The comparison of the reliability of the two airplanes was done by computing the probabilities of mission success and aircraft safety. As can be seen from

TABLE 41 SUMMARY OF TRADE STUDY RESULTS

TRADE PARAMETERS	BASELINE AIRPLANE			ALL-ELECTRIC AIRPLANE		
	ACTUATION	SECONDARY POWER	OVERALL	ACTUATION	SECONDARY POWER	OVERALL
WEIGHT	1165 LBS	1202 LBS	2367 LBS	1418 LBS	944 LBS	2362 LBS
MISSION SUCCESS PROBABILITY			0.995608			0.995289
AIRCRAFT SAFETY PROBABILITY			0.999868			0.999864
MTBF	139 HRS	67 HRS	45 HRS	46 HRS	102 HRS	32 HRS
LCC (500 SHIPSETS)	$83.1M	$71.5M	$154.6M	$96.5M	$41.6M	$138.1M
(1000 SHIPSETS)	$139.0M	$120.8M	$259.8M	$162.1M	$70.9M	$233.0M

Table 41 the results were quite similar in both cases. The measure of maintainability was evaluated by computing the mean-time-between-failures (MTBF) for the two airplanes. The MTBF for the hydraulic actuation systems was almost three times higher than for the EM actuation systems. However, the MTBF of the secondary power system for the All-Electric Airplane was about 50% higher than that of the Baseline Airplane. This resulted in the overall Baseline Airplane secondary power and actuation systems MTBF being 33% higher than that for the All-Electric Airplane.

The life cycle cost for EM actuation systems was 16% higher than the hydraulic actuation systems. On the other hand, the LCC cost of the secondary power system for the All-Electric Airplane was 42% lower than the Baseline Airplane secondary power system. This resulted in the overall LCC of the All-Electric Airplane being approximately 12% less than the Baseline Airplane.

In addition to the quantitative analysis, the systems of the two airplanes were evaluated with respect to six other parameters on a qualitative basis. A summary of this comparison is shown in Table 42.

The fact that electrical systems are designed for twice the maximum average load capacity allows additional growth advantage in the All-Electric Airplane secondary power system over the Baseline Airplane. From a survivability/vulnerability standpoint, hydraulic actuation (where linear pistons are used) is better than EM actuation since the simplicity of design of the hydraulic ram actuators precludes the possibilities of jamming that may occur in lightweight gearboxes used on EM actuators. Electrical power systems have the capability of isolating an individual circuit which has failed and shorted through the action of circuit breakers. Similarly, hydraulic systems can be fused and isolation provided to maintain system integrity should a hydraulic line be broken or damaged, especially due to weapons effects. Aircraft fires can be fueled by leakage of hydraulic fluid. MIL-H-5606 was used for weight estimating purposes in this study. Fire resistant hydraulic fluid, currently under development, is heavier and would add a weight penalty to the hydraulic system.

The All-Electric Airplane will be more vulnerable to the electromagnetic

TABLE 42 COMPARISON OF RELATED FACTORS

EVALUATION PARAMETERS	BASELINE AIRPLANE SECONDARY POWER SYSTEM	BASELINE AIRPLANE ACTUATION SYSTEM	ALL ELECTRIC AIRPLANE SECONDARY POWER SYSTEM	ALL ELECTRIC AIRPLANE ACTUATION SYSTEM
STRUCTURAL INTEGRATION	• LIMITED AMAD LOCATION	• LINEAR ACTUATORS REQUIRE OUT-OF-CONTOUR FAIRINGS	• GENERATORS MOUNTED ON ENGINE SPINNER • LOCATION OF CONDITIONING EQUIPMENT IS FLEXIBLE	• EMA PROVIDES MORE FLEXIBILITY
GROWTH	• ELECTRICAL SYSTEM SIZED FOR GROWTH • HYDRAULIC SYSTEM SIZED TO REQUIREMENTS	• ACTUATORS SIZED TO REQUIREMENTS	• ELECTRICAL SYSTEM SIZED FOR GROWTH	• ADDITIONAL CAPABILITY AVAILABLE FOR SHORT DURATIONS
SURVIVABILITY/ VULNERABILITY	• TOTAL SEPARATION AND ISOLATION TO PREVENT LOSS OF MULTIPLE SOURCES	• SEPARATION OF REDUNDANT ACTUATORS REQUIRED FOR PRIMARY CONTROLS • JAM TOLERANT CAPABILITY • FLUID FIRES	• FAULT ISOLATION CAPABILITY • SYSTEM INTEGRITY AFFECTED BY BUS TIE CAPABILITY	• EMA SUSCEPTIBLE TO CATASTROPHIC JAM
EMC/LIGHTNING	• HYDRAULICS NOT VULNERABLE • ELECTRICAL SYSTEM REQUIRES PROTECTION	• FLY-BY-WIRE SYSTEMS REQUIRE PROTECTION	• ELECTRICAL SYSTEM REQUIRES PROTECTION	• FLY-BY-WIRE SYSTEMS REQUIRE PROTECTION • EM ACTUATION SYSTEMS REQUIRE PROTECTION
ENVIRONMENTAL CONSTRAINTS	• NOT CRITICAL	• NOT CRITICAL	• CONDITIONING EQUIPMENT REQUIRES COOLING	• LOCATION OF EMA CONTROLLERS LIMITED DUE TO COOLING REQUIREMENTS
TECHNOLOGY RISKS	• NO MAJOR RISKS	• NO MAJOR RISKS	• HVDC STARTER/GENERATOR AND POWER DISTRIBUTION • INTEGRITY OF REDUNDANT POWER BUS	• JAM RESISTANT PRIMARY FLIGHT CONTROL EMA • GEARBOX DESIGN • ACTUATION EFFICIENCY

threats due to electromagnetic interference (EMI) and lightning, especially since future aircraft will be utilizing more and more non-metallic (fiberglass, composite) structures.

The All-Electric Airplane is also penalized if the EM actuation and electrical power systems have to operate in an ambient where high temperatures may exist. The distributed hydraulic system has the advantage of using fluid to remove heat from the actuators which can then be transferred by means of heat exchangers to a suitable sink such as the fuel. However, on dead-ended systems, or those that are inactive during flight, such as the landing gear actuation systems, thermal problems do occur (both overheating and freezing on some missions) so special protective measures may be required. The systems used on the Baseline Airplane are a projection of a technology that has a high probability of being achieved. In the All-Electric Airplane the projected technology is higher risk with developments required in the use of high voltage DC, gearbox and motor design, electrical power integrity, actuation, redundancy management, and survivability of control designs.

7.2 Conclusions

Based on the results of this study it is concluded that an All-Electric Airplane is feasible assuming that appropriate development is pursued. For an airplane of the size and mission as that studied in this program, the weight and the reliability/maintainability factors are about equal. A reduction in life cycle cost in the secondary power system can be achieved by extracting a single type of power (electrical) rather than by extracting two types (electrical and hydraulic). Moreover, this reduction is not only adequate to make up for the increase in EM actuation LCC but also to provide a net overall reduction over the Baseline Airplane.

The other six factors that were considered provided advantages and disadvantages for both aircraft designs that offset each other to some extent. Efforts to improve on the hydraulic actuation and hydraulic power systems are continuously being pursued by the military, aircraft manufacturers, and systems vendors. Certain problems associated with EM actuation and electrical power systems are also being pursued. For example, development of EM actuators is being pursued by the same set of agencies listed above. One area of concern is the high risk associated with the use of light weight gearboxes in EM actuators, especially for primary flight control actuation. A lower risk alternative is the integrated actuator package (IAP) which can be utilized in the most critical applications of the primary flight controls, thus allowing the achievement of an All-Electric Airplane.

Another area of concern is the vulnerability of fly-by-wire/power-by-wire systems to electromagnetic threats due to EMI and lightning. Work is being done to devise methods to protect the electrical/ electronic equipment, without undue cost and weight penalties. The F-16 is truly a FBW airplane. As is always the case with radical departure from tried and true methods, the F-16 has had its problems, but none that can be called insurmountable. The FBW electronics are probably more vulnerable to EMC/lightning effects than the PBW or EM actuation systems, since the latter are operating at much higher power levels and hence are less likely to be impacted by electromagnetic noise or transients. In any case, shielding techniques are being developed that are expected to provide the necessary protection for both electronics LRUs and actuators.

Considerable effort was expended in this study to ensure that the EM actuation systems would not be subjected to excessive temperatures during supersonic operations. For subsonic aircraft the additional cooling provisions for the EM actuation system controllers could be reduced considerably or even eliminated. This would result in reduction of the EM actuation system weight, improvement in MTBF and further reduction in LCC. A comparable cooling system requirement does not exist in the Baseline Airplane hydraulic actuation system so the mission change would not provide a comparable reduction.

It is also anticipated that for a much larger aircraft the weight differential in the secondary power system could be greater. This would be possible because of the relatively greater increase in the weight of hydraulic tubing and fittings and the hydraulic fluid in the system. This would also result in additional LCC reductions in the All-Electric Airplane.

VIII RECOMMENDATIONS

This study was based on the premise that certain technology needs in the EM actuation and electrical power systems will be fulfilled. These are:

- o Higher energy product Sm-Co permanent magnets
- o More efficient power switches
- o Better heat removal techniques
- o More efficient and lighter weight gearboxes and ballscrews
- o Protection of PBW electronics from electromagnetic threats
- o Development of optimum type of electrical power generation and distribution system

Also to be fulfilled are technology needs in the hydraulic actuation and power systems as follows:

- o Higher pressure hydraulics
- o Advanced hydraulic components
- o Special hydraulic actuation components
- o Fire resistant hydraulic fluids

Therefore, it is recommended that developments in the following areas be pursued:

<u>For Baseline Airplane</u>

- o Actuation Systems
  - load adaptive/stored energy actuation
  - staged sequential servo ram actuation

- o Advanced Hydraulic Systems
  - high pressure pumps, seals, tubing, and fittings
  - bidirectional power transfer units
  - hydraulic fuses and circuit breakers

- o Fire Resistant Hydraulic Fluid

For All-Electric Airplane

- o Gearboxes
  - light weight
  - high efficiency
  - jam resistant/tolerant

- o Motors
  - length/diameter/power/inertia/speed parametric data

- o Motor/Gearbox Optimization Techniques
  - speed optimized for maximum power transfer

- o Load Adaptive/Stored Energy Actuation Techniques

- o Controller/Inverters
  - thermal management
  - EMC/lightning protection
  - multiplex data bus interface

- o High Voltage DC Electric Systems
  - starter/generator
  - power switching/protection/distribution
  - EMC/lightning protection

In addition to the above it also is recommended that developments in the following areas be pursued since they will be applicable to both airplane types.

- o Integrated Actuator Packages
  - Servo Pump Concept
  - Servo Valve/Accumulator Concept
  - Fixed Displacement Pump Concept

- o Gearless Speed Reduction Motors

- Electromechanical Brake System

- Closed Loop Environmental Control Systems

Developments suggested above would help to provide the technical basis to allow the option of selecting the best solution to optimize the particular airplane configuration and design being considered.

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8. Contract F33615-79-C-2006 "Protection of Advanced Electrical Power Systems from Atmospheric Electromagnetic Hazards."

9. Report AFFDL-TR-76-42, "Electromechanical Actuation Feasibility Study," AiResearch Manufacturing Company, May 1976.

10. AFWAL-TR-81-2012, "Aircraft Digital Input Controlled Hydraulic Actuation and Control System," Boeing Military Airplane Company, March 1981.

APPENDIX A

RELIABILITY DATA

This appendix contains the mission loss fault trees and the airplane loss fault trees for both the Baseline and the All-Electric Airplanes. In addition, computer printouts are included for probability of mission completion and probability of airplane safety for both the Baseline and the All-Electric Airplanes.

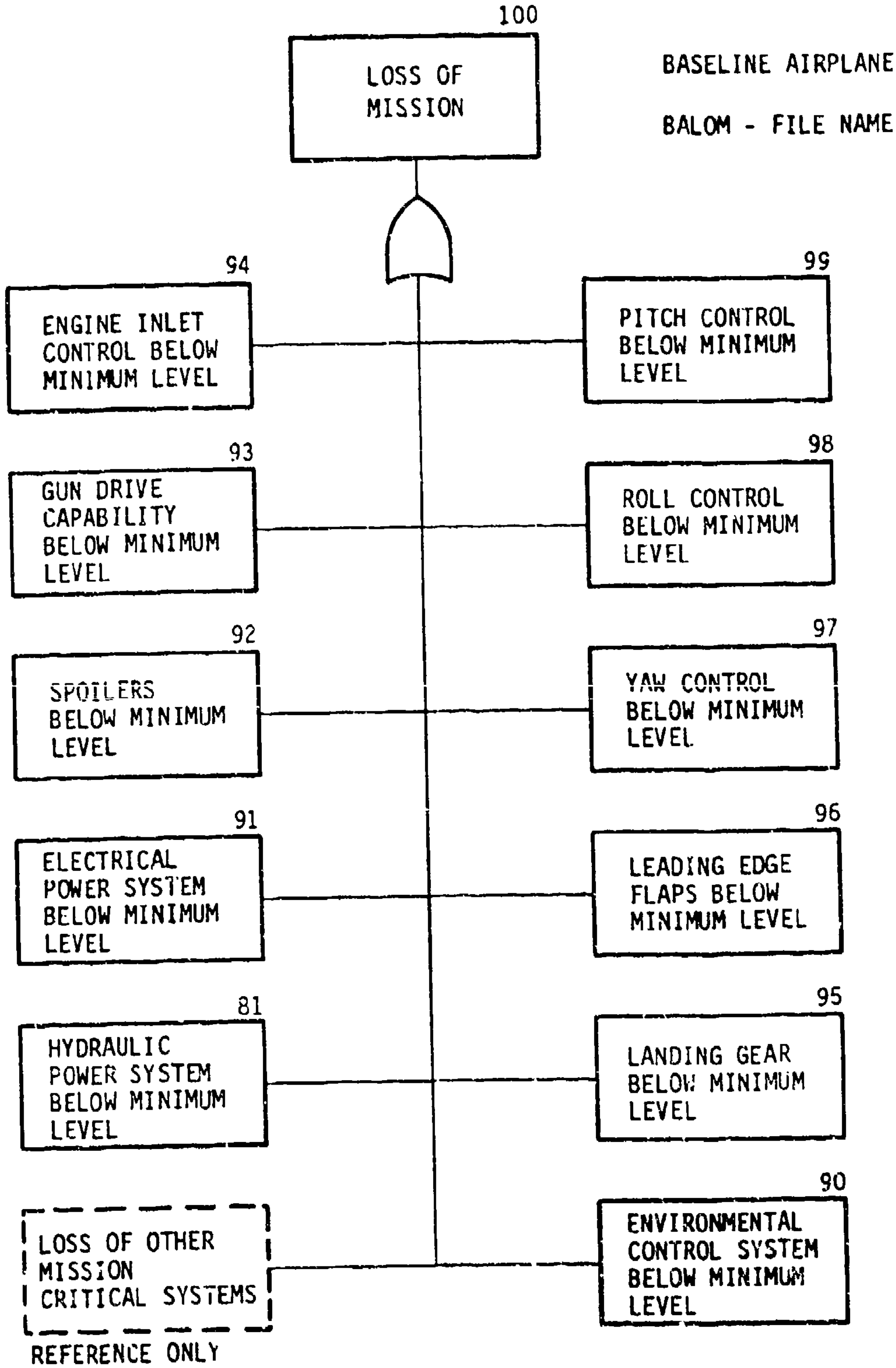
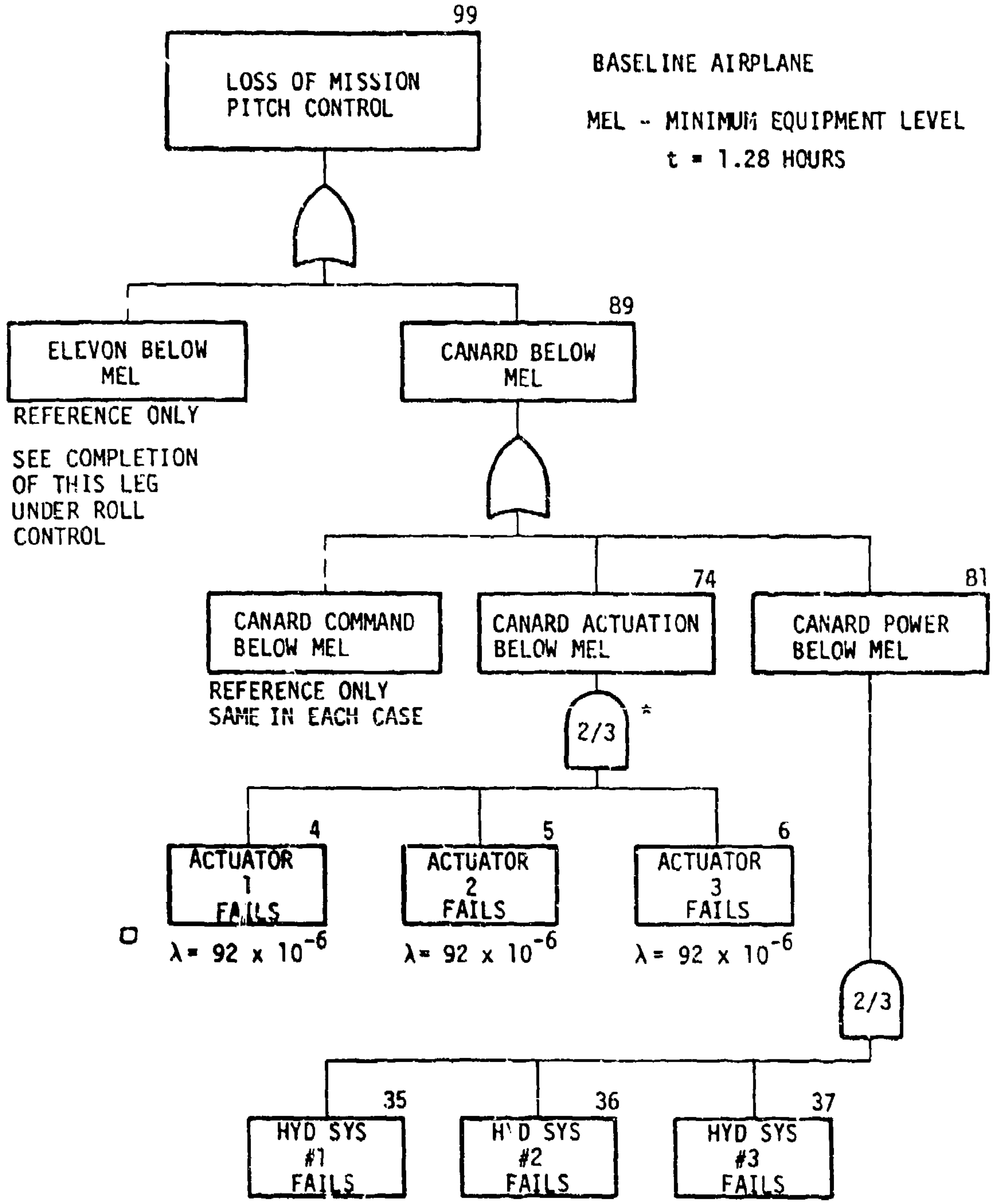
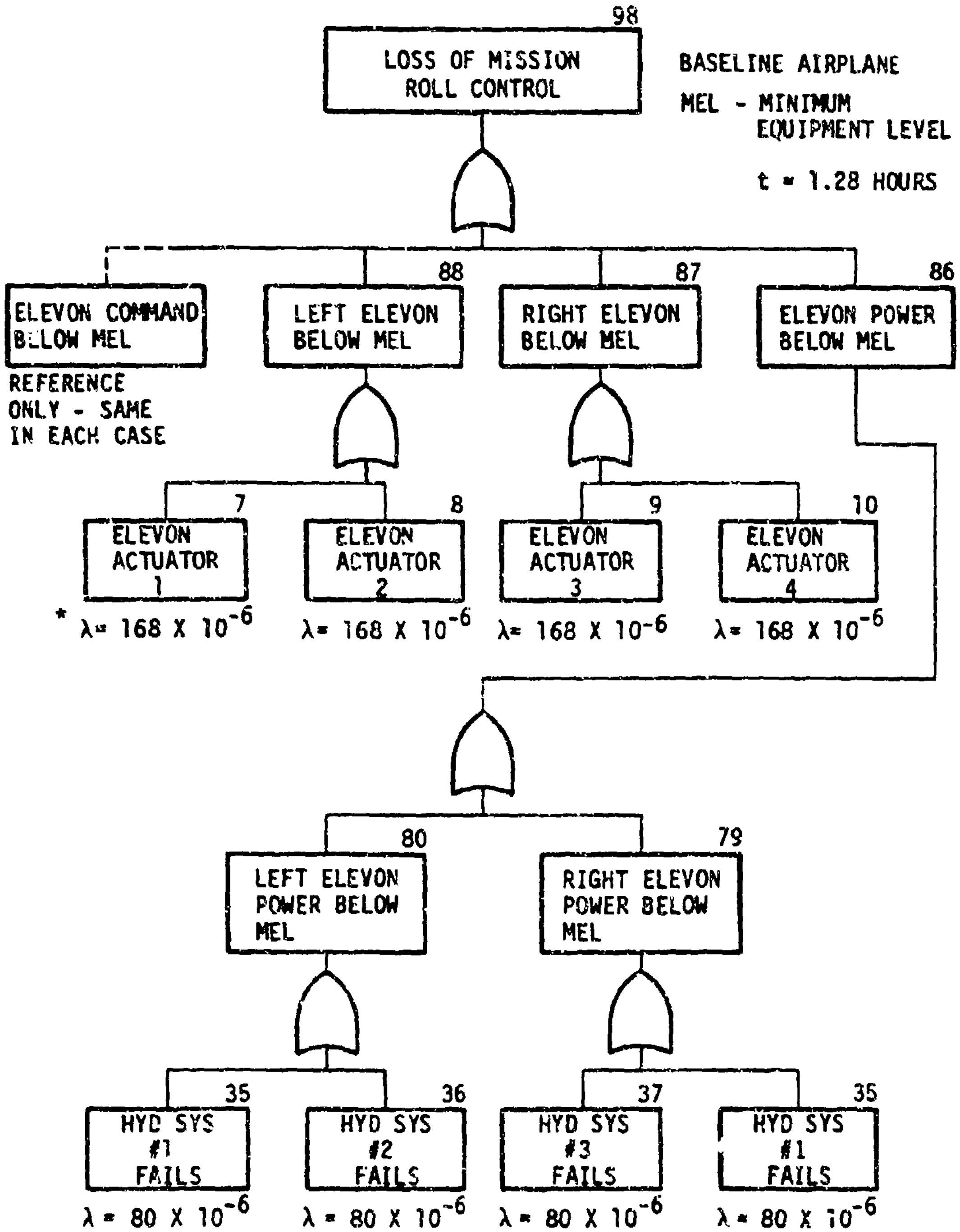


Figure A-1 Mission Fault Tree for Baseline Airplane



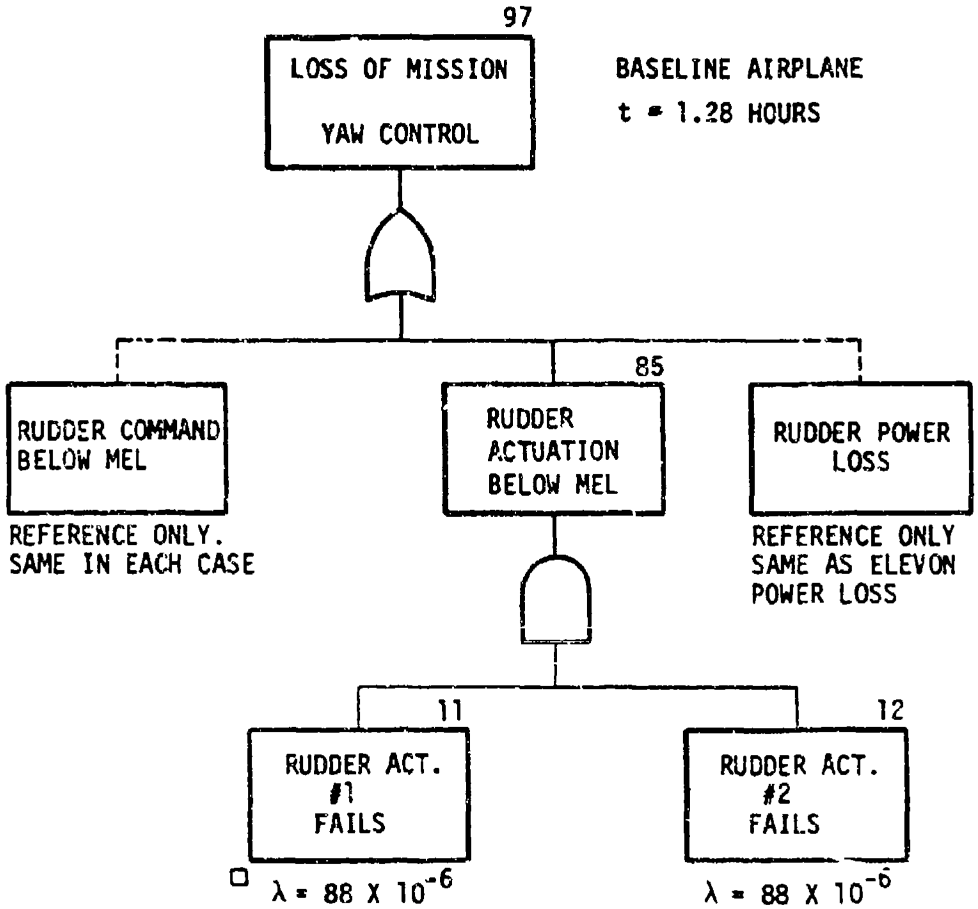
* TWO-DUAL TANDEM ACTUATORS CONTROLLING CANARD DEFINED AS THREE REDUNDANT ACTUATORS
** FAILURE OF EITHER HYD SYSTEM CAUSES MISSION ABORT UNDER ELEVON FAULT TREE
□ C-14 ELEVATOR PCU FR X 2

Figure A-2 Loss of Mission Fault Tree - Pitch Control Baseline Airplane



C-14 FR FOR AILERON PCU
+ CONTROL VALVE MODULE
X 2 FOR FIGHTER ENVIRONMENT

Figure A-3 Loss of Mission Fault Tree - Roll Control Baseline Airplane



□ C-14 RUDDER PCU FR X 2

Figure A-4 Loss of Mission Fault Tree - Yaw Control Baseline Airplane

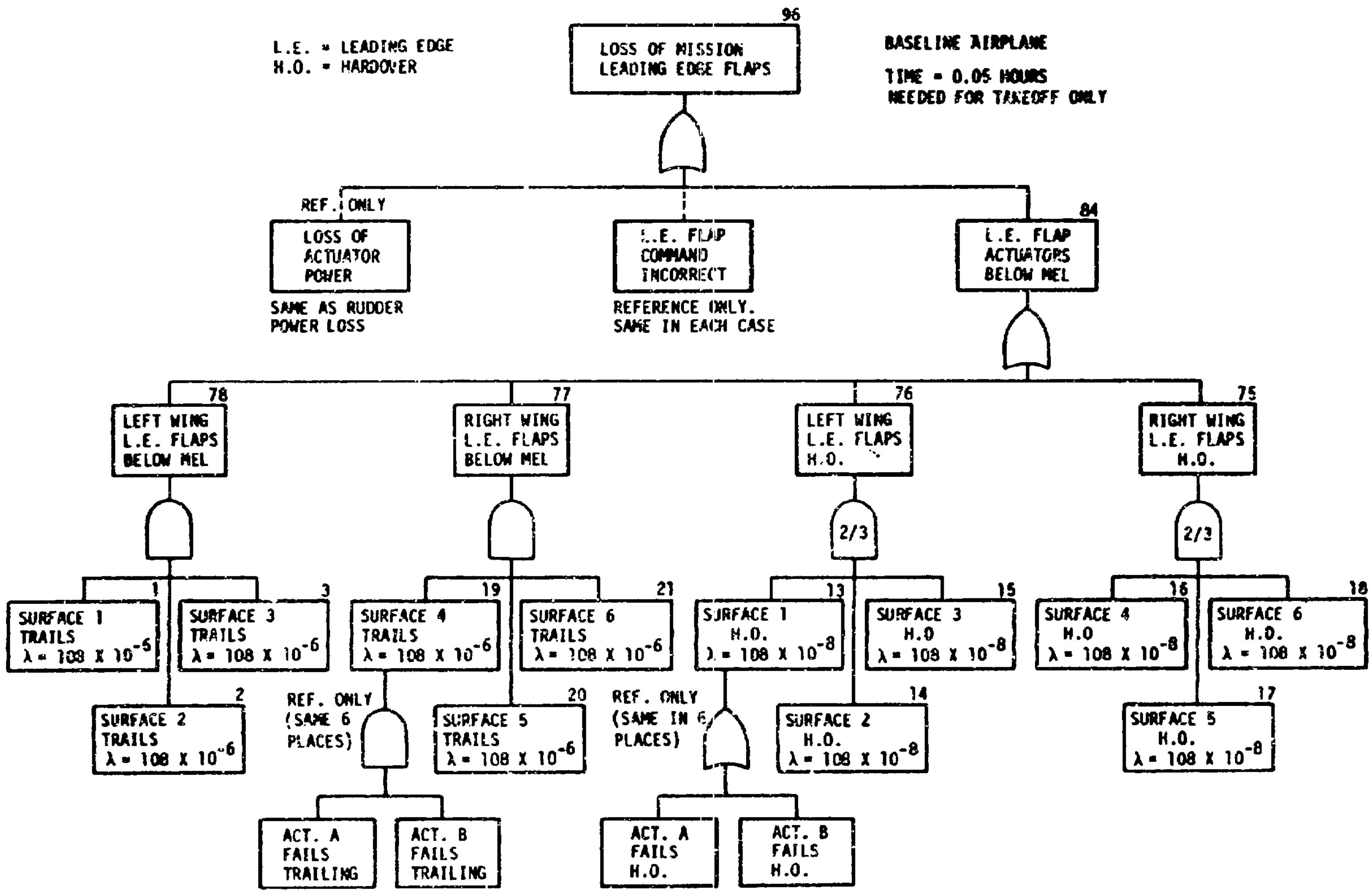
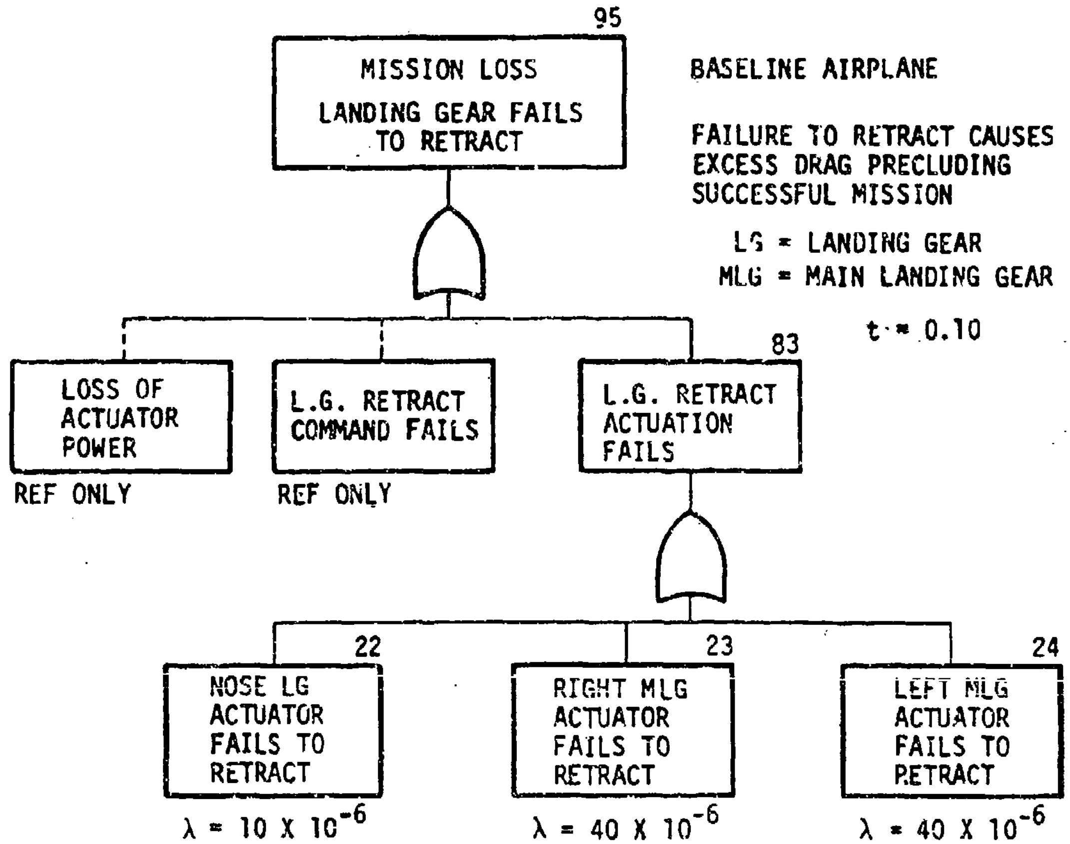


Figure A-5 Loss of Mission Fault Tree - L.E. Flaps Baseline Airplane



- NOSE GEAR RETRACTS FORWARD THEREFORE CAN EXTEND BY FREE-FALL

- INADVERTENT EXTENSION IS CONSIDERED TO BE AN IMPOSSIBLE FAILURE MODE

Figure A-6 Loss of Mission Fault Tree - Landing Gear Baseline Airplane

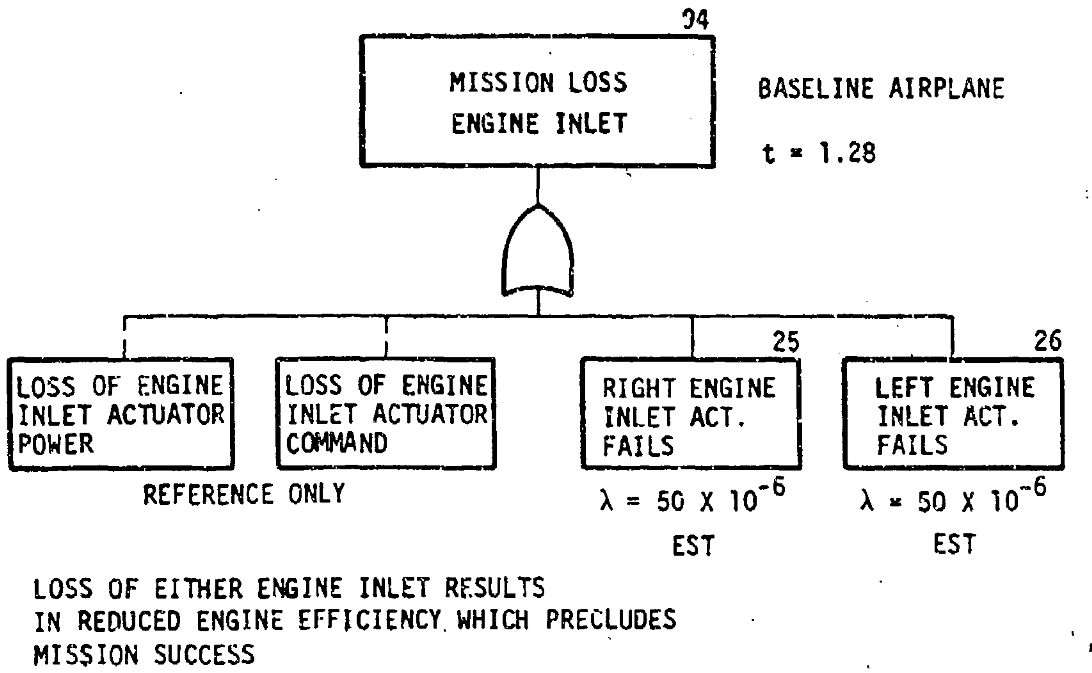


Figure A-7 Loss of Mission Fault Tree - Engine Inlet Baseline Airplane

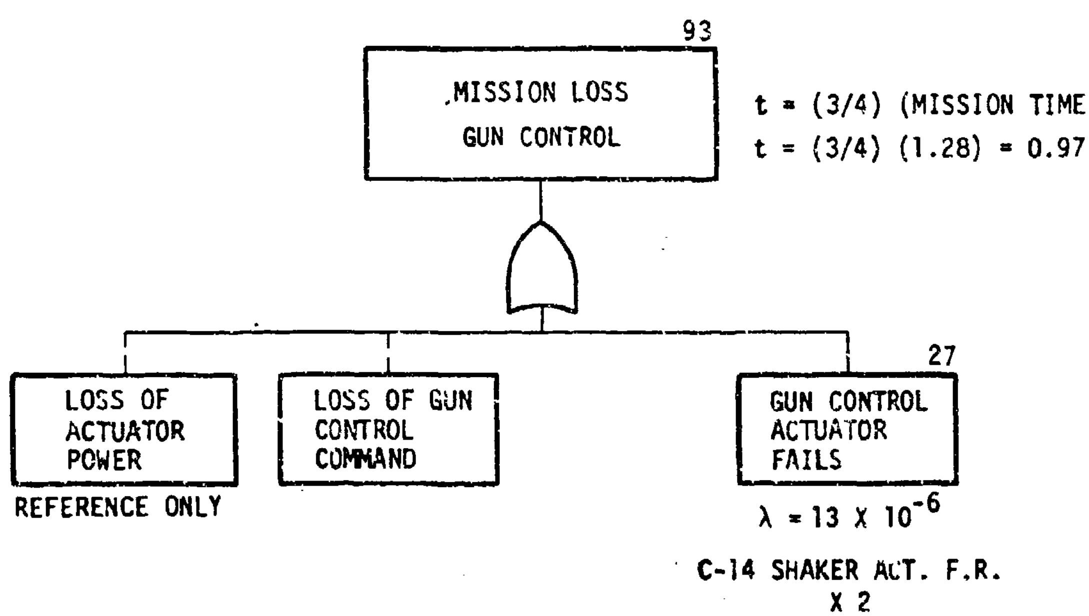


Figure A-8 Loss of Mission Fault Tree - Gun Control Baseline Airplane

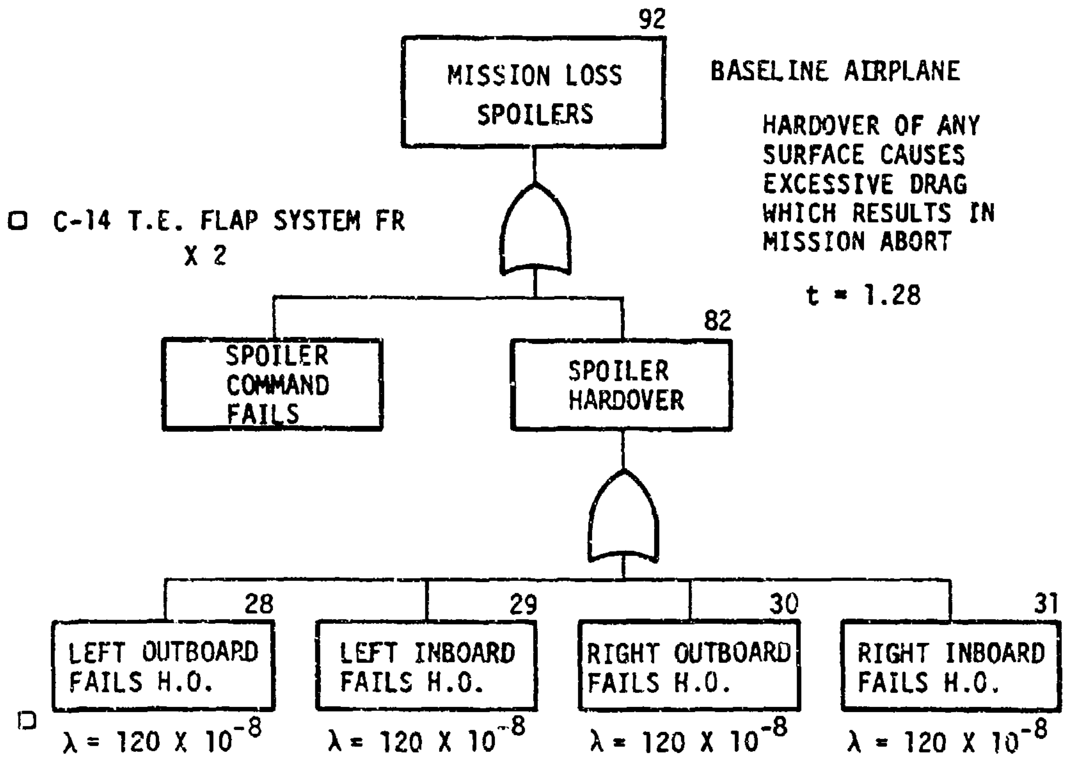
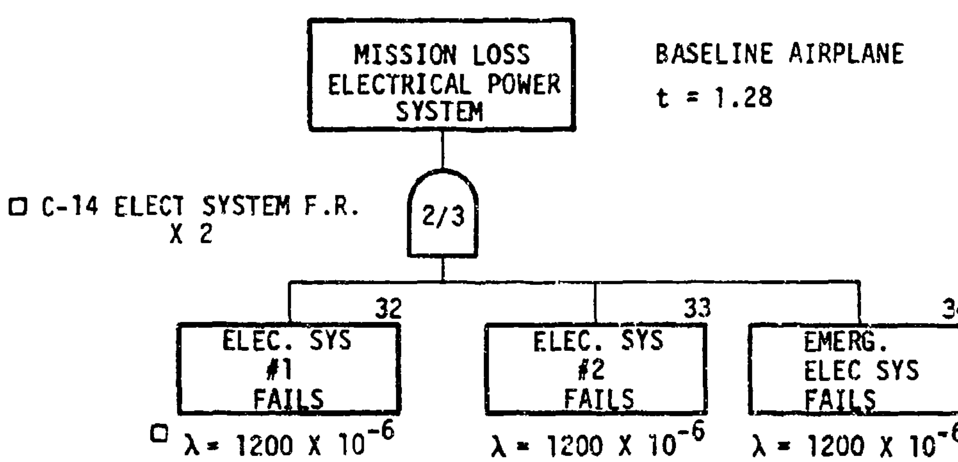
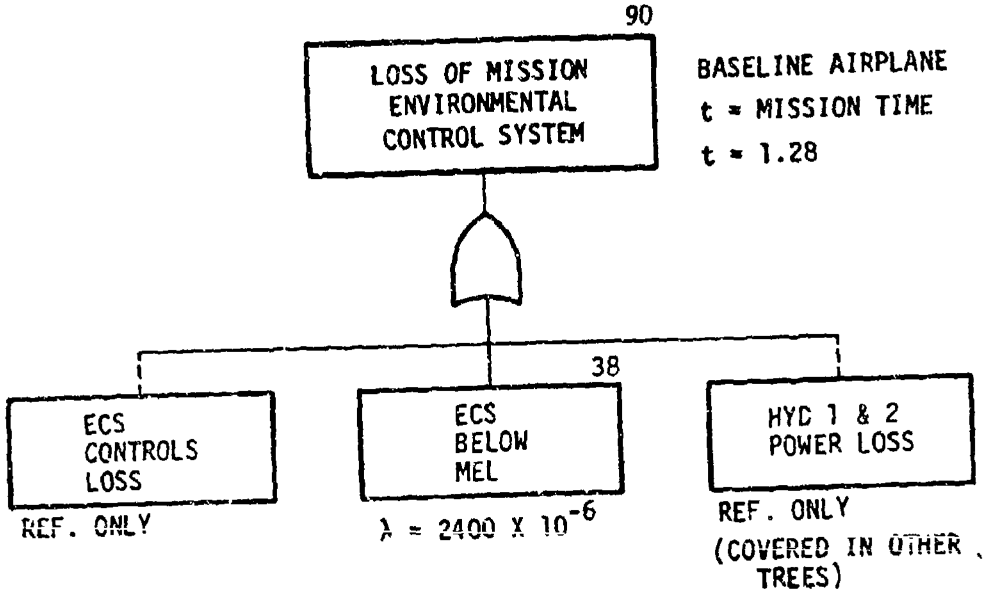


Figure A-9 Loss of Mission Fault Tree - Spoilers Baseline Airplane



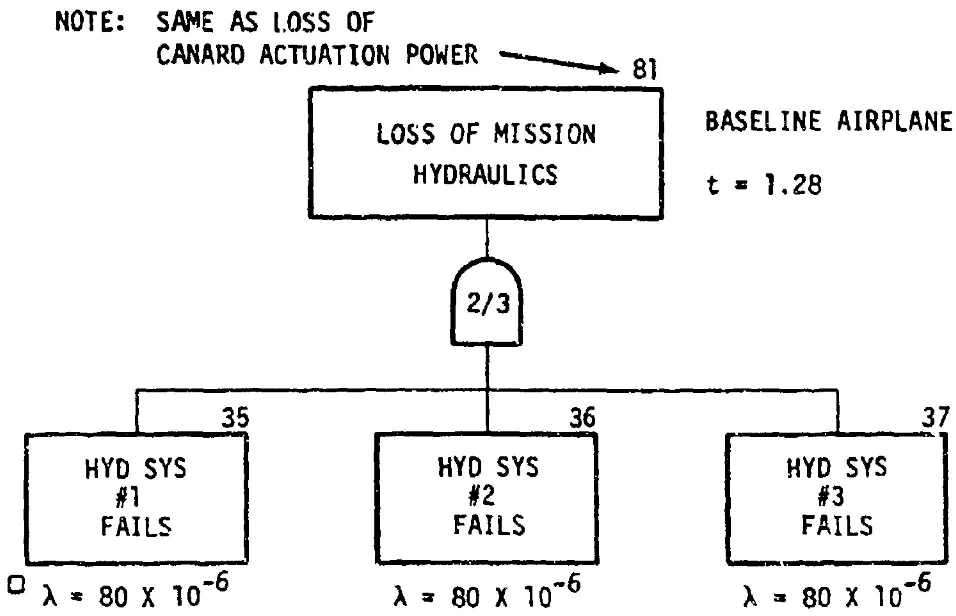
- LOSS OF 2 OF 3 ELECTRICAL SYSTEMS RESULTS IN MISSION ABORT.
- EACH "SYSTEM" ASSUMED TO CONTAIN ITS OWN DISTRIBUTION SYSTEM
- ASSUMES NO SINGLE FAILURE POINTS EXIST THAT CAN CAUSE ALL SYSTEMS TO GO DOWN AT ONCE.
- IGNORES LOSS OF ENGINES AS A CAUSE OF ELECTRICAL SYSTEM LOSS SINCE THE EFFECTS ARE THE SAME FOR BOTH BASELINE & ALL-ELECTRIC AIRPLANES

Figure A-10 Loss of Mission Fault Tree - Electrical Power System Baseline Airplane



□ CECS F.R. FOR ECS (OPEN LOOP) = 2427×10^{-6}

Figure A-11 Loss of Mission Fault Tree - ECS Baseline Airplane



- LOSS OF HYD SYSTEMS #1 AND #2 CAUSES LOSS OF AERIAL REFUEL, GUN DRIVE, AND ECS WHICH RESULTS IN MISSION ABORT. THIS ASSUMES THAT THE ECS FAILURE IS DETECTABLE AND ABORT CAN BE ACCOMPLISHED BEFORE LOSS OF CRITICAL FLY-BY-WIRE AVIONICS OCCURS, OTHERWISE LOSS OF A/C CAN RESULT FROM LOSS OF BOTH HYDRAULIC SYSTEMS.
- EACH "SYSTEM" ASSUMED TO CONTAIN ITS OWN DISTRIBUTION SYSTEM.
- ASSUMES NO SINGLE FAILURE POINTS EXIST THAT CAN CAUSE ALL SYSTEMS TO GO DOWN AT ONCE.
- IGNORES LOSS OF ENGINES AS A CAUSE OF HYD SYSTEM LOSS

□ C-14 HYD SYS F.R. X 2

Figure A-12 Loss of Mission Fault Tree - Hydraulic Power System Baseline Airplane

TABLE A-1 MISSION COMPLETION - BASELINE AIRPLANE

BASELINE AIRPLANE - [illegible] MISSION [illegible]

SECTION [illegible] SUMMARY OF THE SYSTEM PROBABILITIES.

[illegible] = 1.[illegible]E+00 [illegible] = 5.000E-02
= 3.[illegible]E-01 [illegible] = 4.500E-01

EVENT	UNRELIABILITY	RELIABILITY	GATE TYPE	INPUTS OR GATE INPUT	RATE			
1	5.400E-06	0.99999460000145800	I	1	1.080E-04	21		
2	5.400E-06	0.99999460000145800	I	1	1.080E-04	21		
3	5.400E-06	0.99999460000145800	I	1	1.080E-04	21		
4	1.177E-04	0.9998822[illegible]	I	1	9.200E-05	11		
5	1.177E-04	0.9998822[illegible]	I	1	9.200E-05	11		
6	1.177E-04	0.9998822[illegible]	I	1	9.200E-05	11		
7	2.150E-04	0.99978499511944 36	I	1	1.650E-04	11		
8	2.150E-04	0.99978499511944 36	I	1	1.650E-04	11		
9	2.150E-04	0.99978499511944 36	I	1	1.650E-04	11		
10	2.150E-04	0.99978499511944 36	I	1	1.650E-04	11		
11	1.136E-04	0.99988[illegible]	I	1	[illegible]	11		
12	1.136E-04	0.99988[illegible]	I	1	[illegible]	11		
13	5.400E-08	0.99999994600000014	I	1	1.080E-06	21		
14	5.400E-08	0.99999994600000014	I	1	1.080E-06	21		
15	5.400E-08	0.99999994600000014	I	1	1.080E-06	21		
16	5.400E-08	0.99999994600000014	I	1	1.080E-06	21		
17	5.400E-08	0.99999994600000014	I	1	1.080E-06	21		
18	5.400E-08	0.99999994600000014	I	1	1.080E-06	21		
19	5.400E-06	0.99999460000145800	I	1	1.080E-04	21		
20	5.400E-06	0.99999460000145800	I	1	1.080E-04	21		
21	5.400E-06	0.99999460000145800	I	1	1.080E-04	21		
22	1.000E-06	0.99999900000049999	I	1	1.000E-05	21		
23	4.000E-06	0.99999600000799999	I	1	4.000E-05	21		
24	4.000E-06	0.99999600000799999	I	1	4.000E-05	21		
25	5.400E-05	0.9999460[illegible]	I	1	5.400E-05	11		
26	5.400E-05	0.9999460[illegible]	I	1	5.400E-05	11		
27	1.349E-05	0.99998650[illegible]745	I	1	1.350E-05	41		
28	1.536E-06	0.99999846400117[illegible]	I	1	1.300E-06	11		
29	1.536E-06	0.99999846400117[illegible]	I	1	1.300E-06	11		
30	1.536E-06	0.99999846400117[illegible]	I	1	1.300E-06	11		
31	1.536E-06	0.99999846400117[illegible]	I	1	1.300E-06	11		
32	1.535E-03	0.9984651790442580	I	1	1.300E-03	11		
33	1.535E-03	0.9984651790442580	I	1	1.300E-03	11		
34	1.535E-03	0.9984651790442580	I	1	1.300E-03	11		
35	1.024E-04	0.99989760052427010	I	1	5.000E-05	11		
36	1.024E-04	0.99989760052427010	I	1	5.000E-05	11		
37	1.024E-04	0.99989760052427010	I	1	5.000E-05	11		
38	1.067E-03	0.9989327127639702	I	1	5.400E-03	11		
74	4.159E-06	0.99999[illegible]11[illegible]	[illegible]	4	5	[illegible]		
75	[illegible]E-15	0.99999999999999912	[illegible]	15	17	19		
76	[illegible]E-15	0.99999999999999912	[illegible]	13	14	15		
77	1.[illegible]E-16	0.99999999999999999	A	19	39	[illegible]		
78	1.[illegible]E-16	0.99999999999999999	A	1	[illegible]			
79	[illegible].048E-04	0.9997[illegible]	[illegible]	[illegible]	[illegible]			
80	[illegible].048E-04	0.9997[illegible]	O	[illegible]	[illegible]			
81	[illegible].145E-06	0.9999999[illegible]490691	[illegible]	[illegible]	[illegible]	[illegible]		
82	[illegible].144E-06	0.99999[illegible]01[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	31	
83	[illegible].000E-06	0.99999[illegible]000040[illegible]	O	[illegible]	[illegible]	34		
84	1.791E-14	0.99999999999999[illegible]	O	[illegible]	[illegible]	[illegible]	[illegible]	
85	1.[illegible]E-09	0.99999999[illegible]31[illegible]	A	11	1[illegible]			
86	[illegible].077E-04	0.99989[illegible]471[illegible]10[illegible]	[illegible]	79	[illegible]			
87	4.300E-04	0.999570012471146 1	O	[illegible]	1[illegible]			
88	4.300E-04	0.9995700124711461	O	[illegible]	[illegible]			
89	[illegible].10[illegible]E-0[illegible]	0.99999999[illegible]	O	74	[illegible]1			
90	[illegible].067E-03	0.99893271[illegible]	O	[illegible]				
91	[illegible]	0.99999[illegible]912	[illegible]	[illegible]	[illegible]	[illegible]		
92	[illegible].144E-06	0.9999[illegible]0158746	O	[illegible]				
93	1.349E-05	0.99998650077[illegible]74[illegible]	O	[illegible]				
94	1.[illegible]E-04	0.9995720[illegible]1916504	O	[illegible]	[illegible]			
95	[illegible].000E-06	0.99999[illegible]	O	[illegible]				
96	1.791E-14	0.99999999999999[illegible]	[illegible]	[illegible]				
97	1.[illegible]E-0[illegible]	0.99999999[illegible]	O	[illegible]				
98	1.[illegible]E-0[illegible]	0.9999[illegible]10[illegible]	O	[illegible]	[illegible]	[illegible]		
99	[illegible]E-0[illegible]	0.99999[illegible]	O	[illegible]				
100	4.[illegible]E-0[illegible]	0.9[illegible]14[illegible]1	O	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
				[illegible]	[illegible]	[illegible]	[illegible]	99

REDUNDANCIES SPECIFIED ARE
35 36

[illegible] FAILURE COMBINATIONS. APPROXIMATION CUTOFFS: 1.00E-13 [illegible]E-16

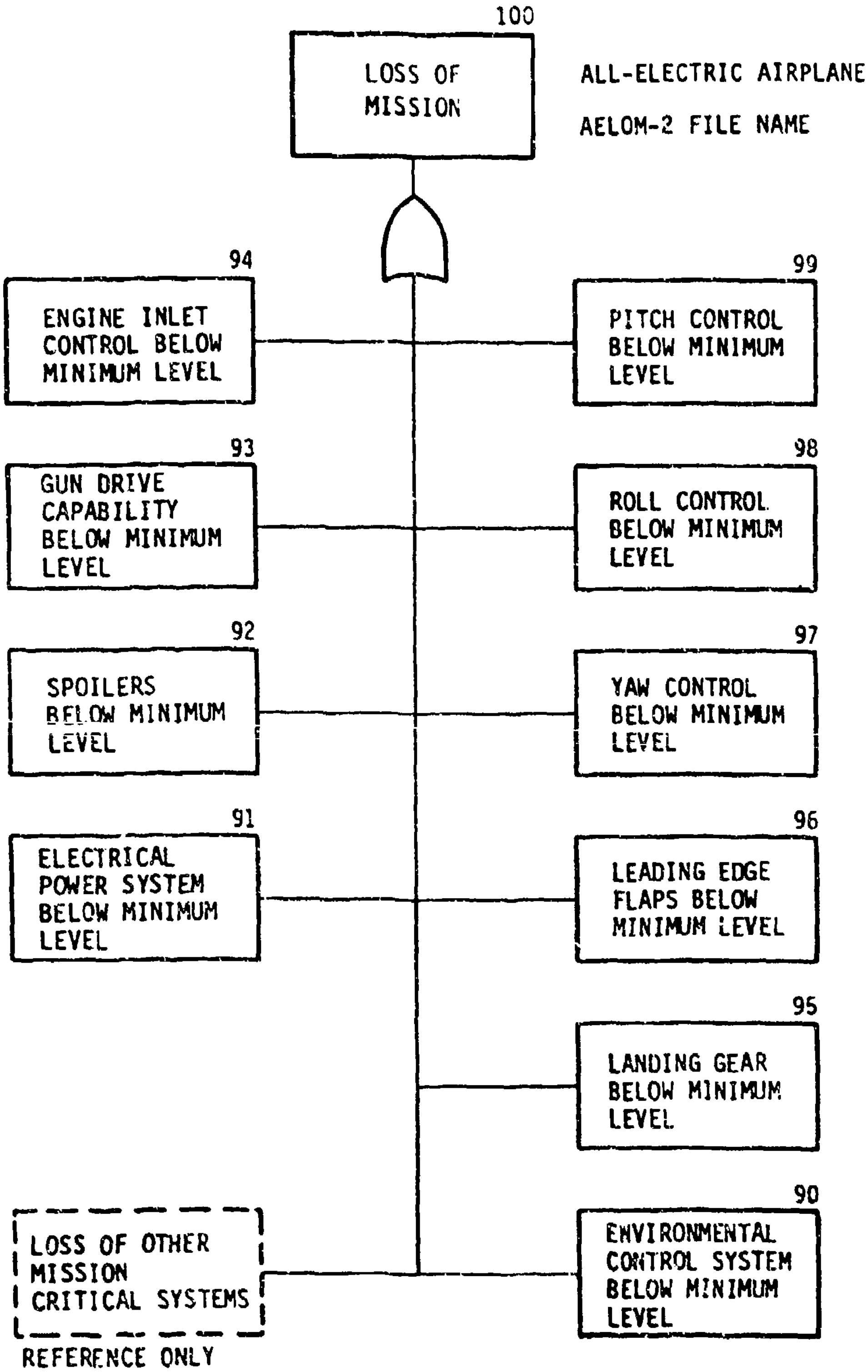


Figure A-13 Loss of Mission Fault Tree - All-Electric Airplane

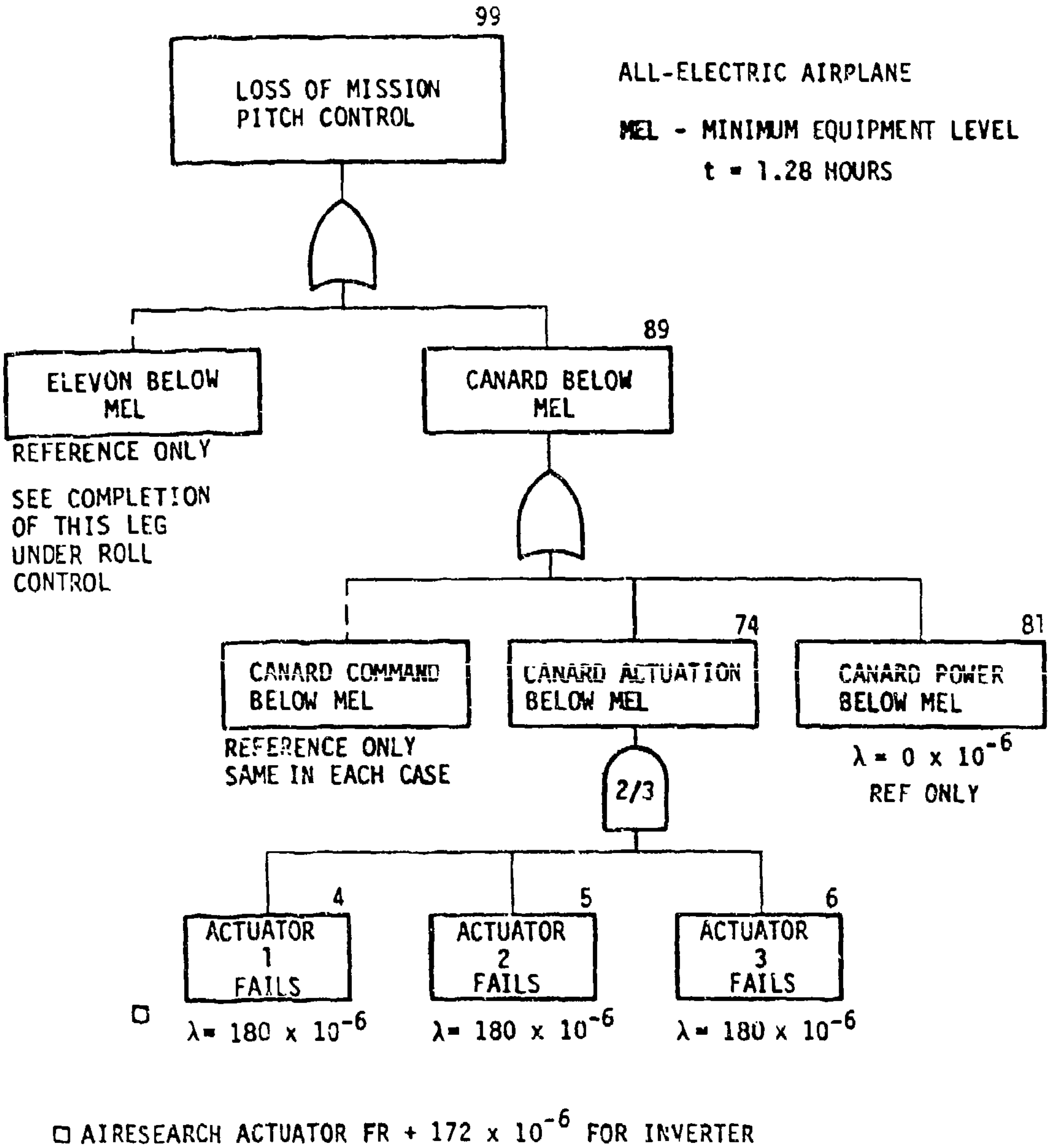
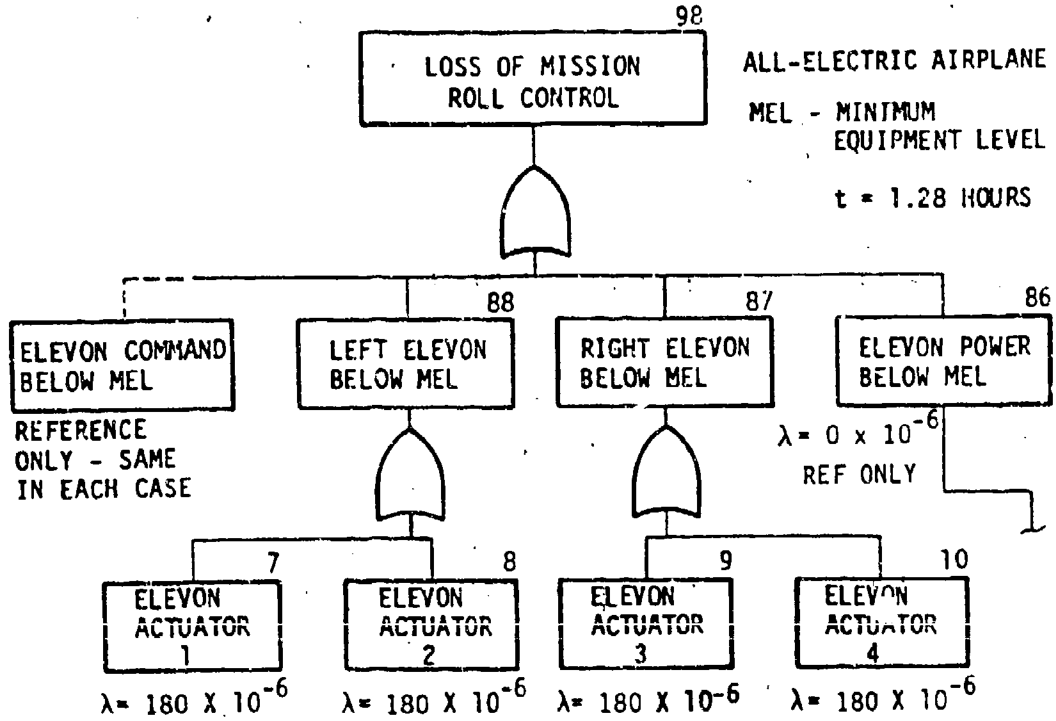


Figure A-14 Loss of Mission Fault Tree - Pitch Control All-Electric Airplane



NOTE:

"ACTUATOR" FAILURE RATE INCLUDES INVERTER AND ACTUATOR

C-14 FR FOR INVERTER $= 86 \times 10^{-6}$

FIGHTER CONVERSION x 2 $= 172 \times 10^{-6}$

AIRESEARCH λ FOR ELEVON ACTUATOR 8.2×10^{-6}

Figure A-15 Loss of Mission Fault Tree - Roll Control All-Electric Airplane

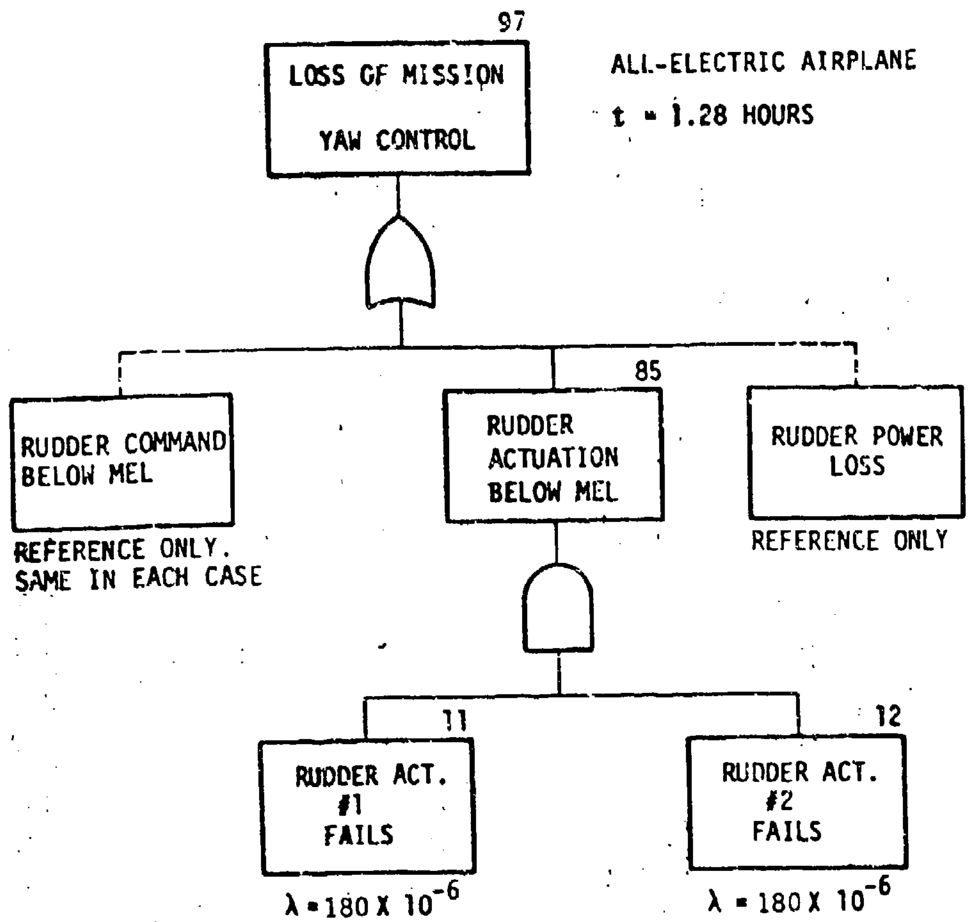


Figure A-16 Loss of Mission Fault Tree - Yaw Control All-Electric Airplane

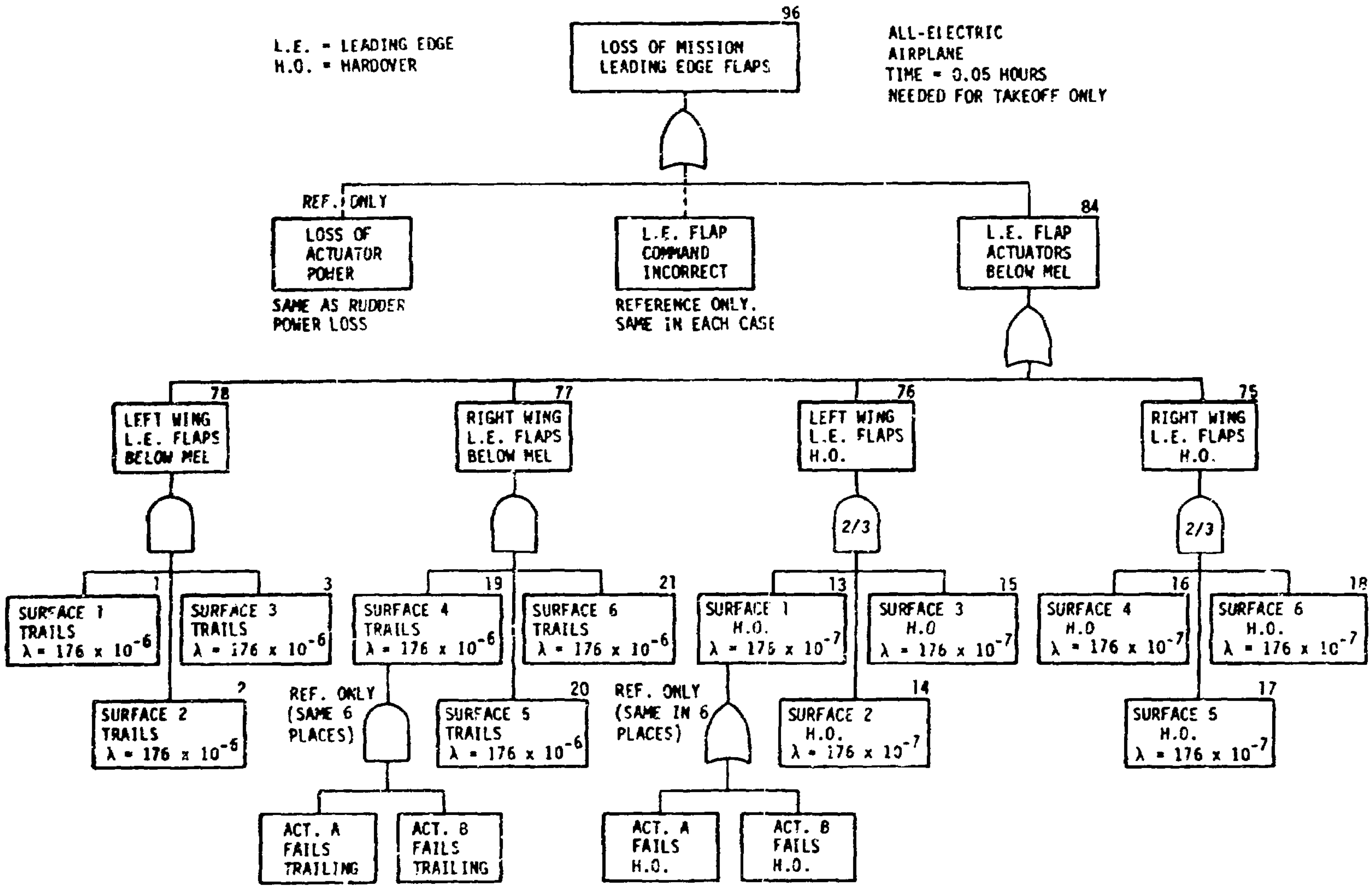
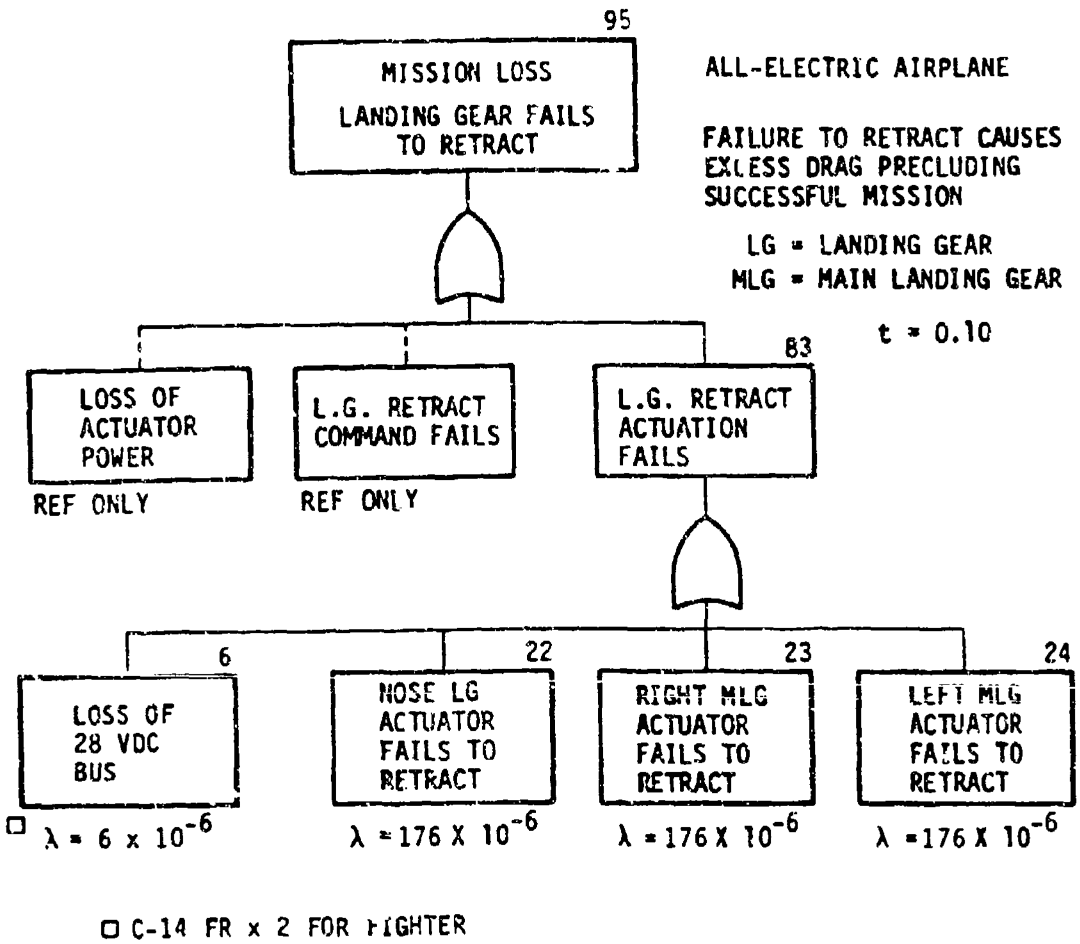


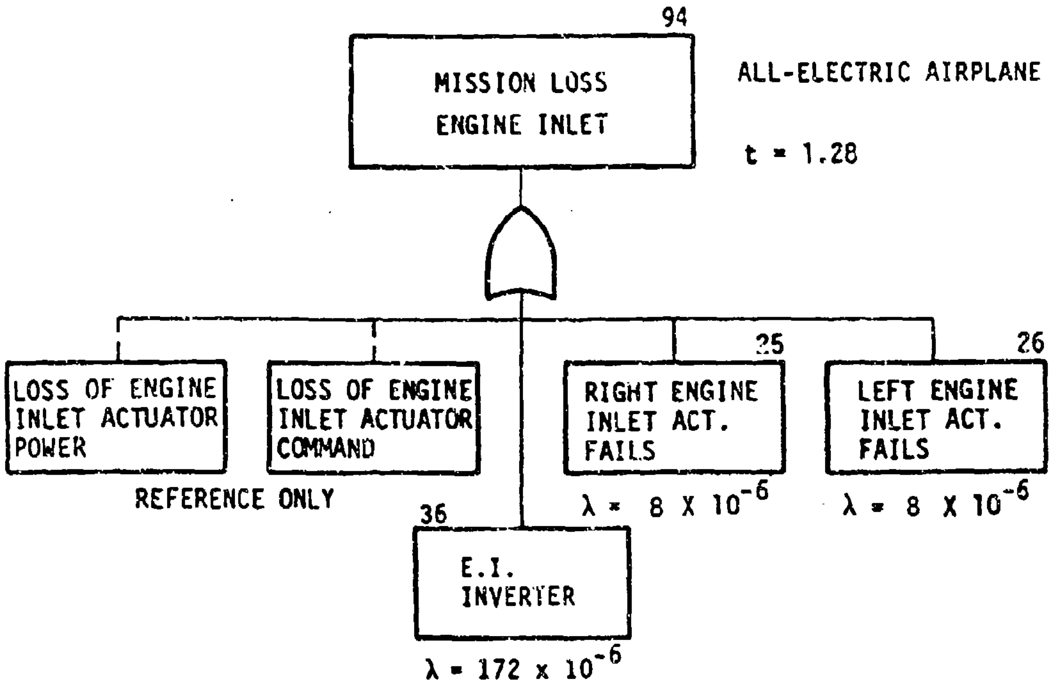
Figure A-17 Loss of Mission Fault Tree - Leading Edge Flaps All-Electric Airplane



- NOSE GEAR RETRACTS FORWARD THEREFORE CAN EXTEND BY FREE-FALL

- INADVERTENT EXTENSION IS CONSIDERED TO BE AN IMPOSSIBLE FAILURE MODE

Figure A-18 Loss of Mission Fault Tree - Landing Gear All-Electric Airplane



LOSS OF EITHER ENGINE INLET RESULTS
IN REDUCED ENGINE EFFICIENCY WHICH PRECLUDES
MISSION SUCCESS

Figure A-19 Loss of Mission Fault Tree -
Engine Inlet All-Electric Airplane

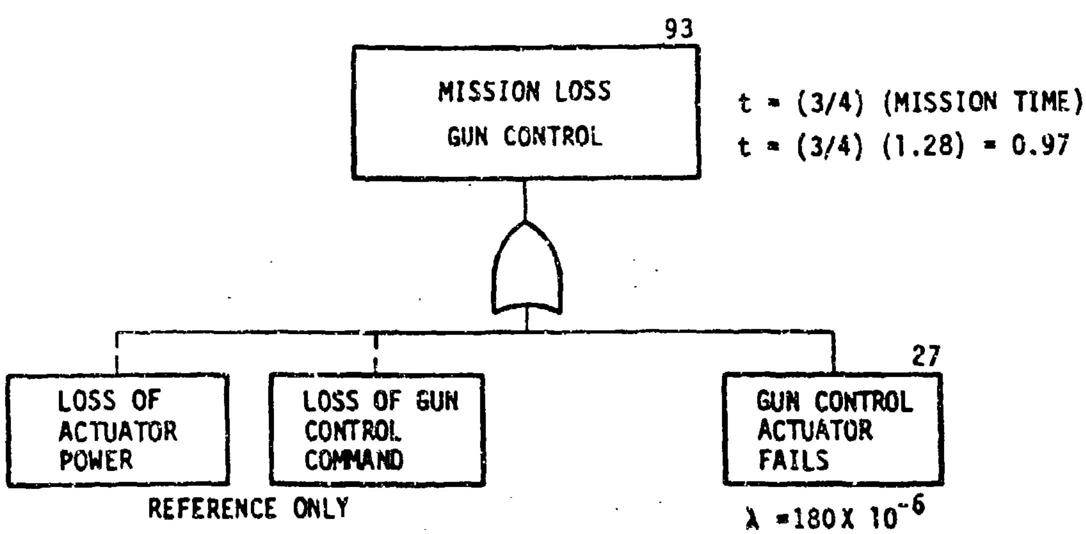


Figure A-20 Loss of Mission Fault Tree -
Gun Control All-Electric Airplane

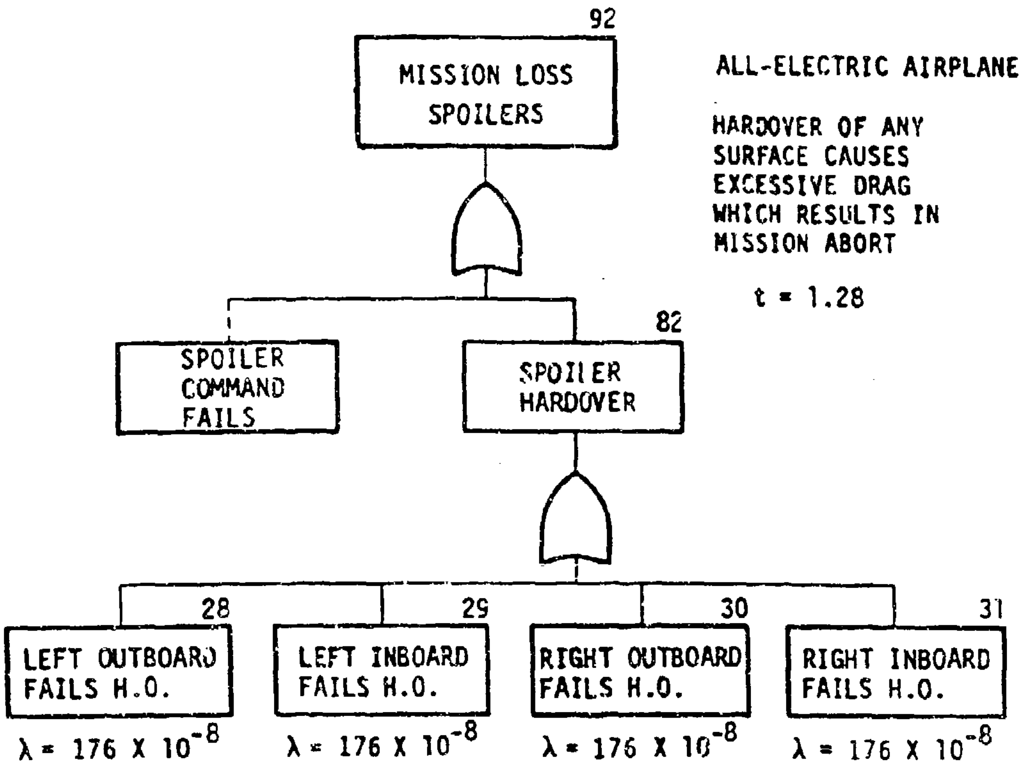
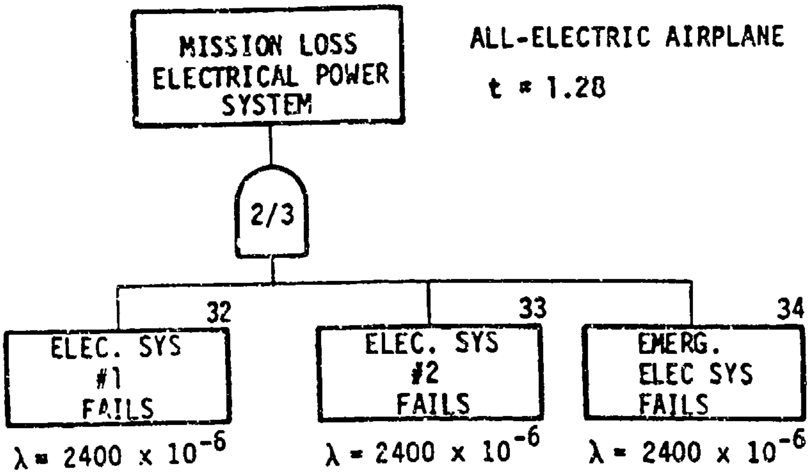


Figure A-21 Loss of Mission Fault Tree - Spoilers All-Electric Airplane



- LOSS OF 2 OF 3 ELECTRICAL SYSTEMS RESULTS IN MISSION ABORT.
- EACH "SYSTEM" ASSUMED TO CONTAIN ITS OWN DISTRIBUTION SYSTEM
- ASSUMES NO SINGLE FAILURE POINTS EXIST THAT CAN CAUSE ALL SYSTEMS TO GO DOWN AT ONCE.
- IGNORES LOSS OF ENGINES AS A CAUSE OF ELECTRICAL SYSTEM LOSS SINCE THE EFFECTS ARE THE SAME FOR BOTH BASELINE & ALL-ELECTRIC AIRPLANES

Figure A-22 Loss of Mission Fault Tree - Electrical Power System All-Electric Airplane

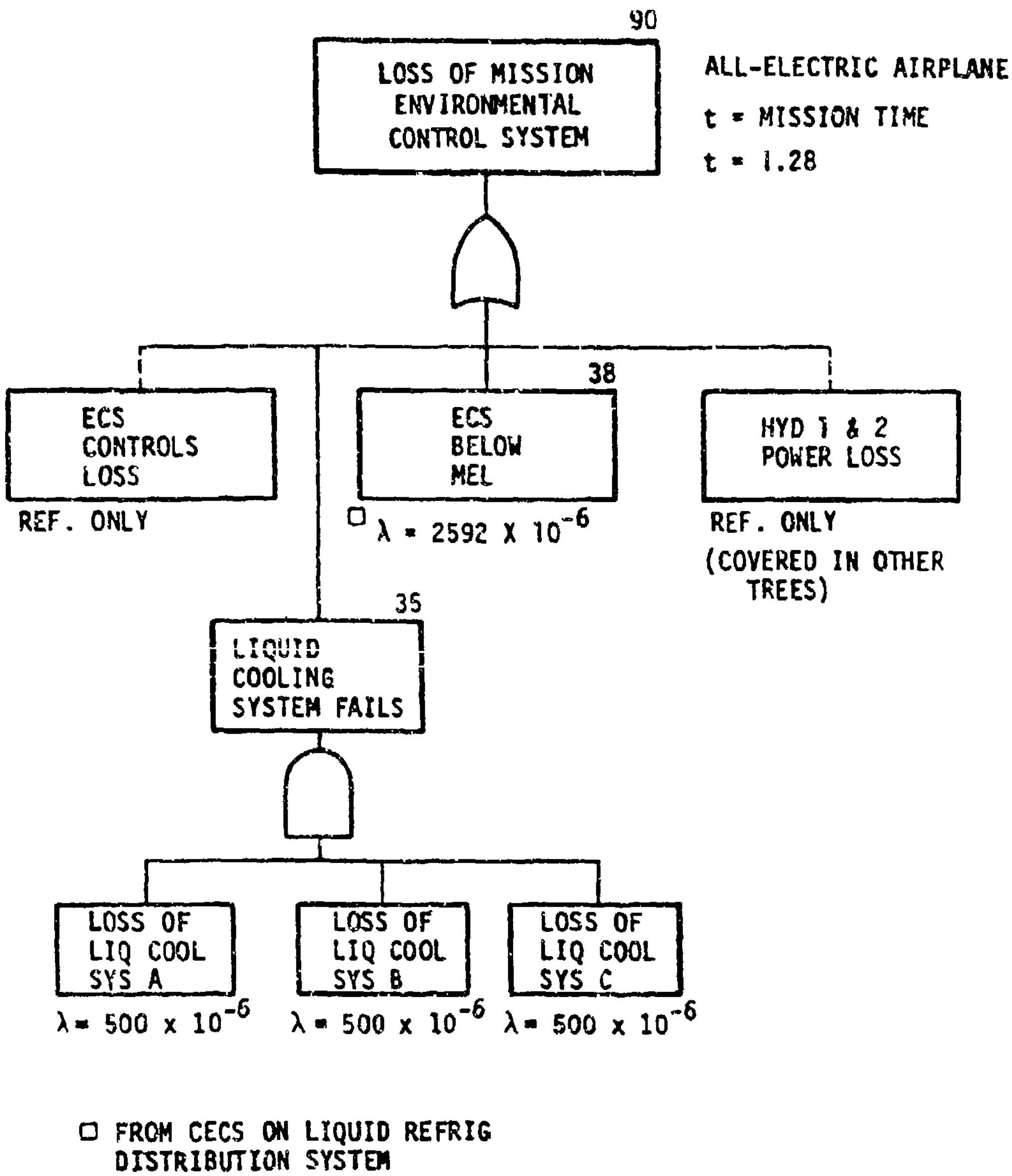


Figure A-23 Loss of Mission Fault Tree - Environmental Control System All-Electric Airplane

TABLE A-2 MISSION COMPLETION - ALL-ELECTRIC AIRPLANE

ALL ELECTRIC AIRPLANE - LOSS OF MISSION [illegible]

OPTION 2, SUMMARY OF THE SYSTEM PROBABILITIES.

FLIGHT HOURS (FH): 1= 1.250D+00 2= 5.000D-02
3= 1.000D-01 4= 9.460D-01

EVENT	UNRELIABILITY	RELIABILITY	GATE TYPE		NUMBER GATE INPUT	
1	8.800D-06	0.99999120000387199	I	1	1.760D-04	[illegible]
[illegible]	8.800D-06	0.99999120000387199	I	1	1.760D-04	[illegible]
[illegible]	8.800D-06	0.99999120000387199	I	1	1.760D-04	[illegible]
4	2.104D-04	0.99978962654004 16	I	1	1.300D-04	11
[illegible]	2.104D-04	0.99978962654004 16	I	1	1.300D-04	11
[illegible]	4.000D-07	0.99999960000001799	I	1	[illegible]	[illegible]
[illegible]	2.104D-04	0.99978962654004 16	I	1	1.500D-04	11
[illegible]	2.104D-04	0.99978962654004 16	I	1	1.500D-04	11
[illegible]	2.104D-04	0.99978962654004 16	I	1	1.500D-04	11
10	2.104D-04	0.99978962654004 16	I	1	1.500D-04	[illegible]
11	2.104D-04	0.99978962654004 16	I	1	1.300D-04	[illegible]
12	2.104D-04	0.99978962654004 16	I	1	1.500D-04	11
13	8.800D-07	0.99999912000003872	I	1	1.760D-05	[illegible]
14	8.800D-07	0.99999912000003872	I	1	1.760D-05	[illegible]
15	8.800D-07	0.99999912000003872	I	1	1.760D-05	[illegible]
16	8.800D-07	0.99999912000003872	I	1	1.760D-05	[illegible]
17	8.800D-07	0.99999912000003872	I	1	1.760D-05	[illegible]
[illegible]	8.800D-07	0.99999912000003872	I	1	1.760D-05	[illegible]
[illegible]	8.800D-06	0.99999120000387199	I	1	1.760D-04	[illegible]
[illegible]	8.800D-06	0.99999120000387199	I	1	1.760D-04	[illegible]
[illegible]	8.800D-06	0.99999120000387199	I	1	1.760D-04	[illegible]
[illegible]	1.760D-05	0.99998240001548790	I	1	1.760D-04	[illegible]
[illegible]	1.760D-05	0.99998240001548790	I	1	1.760D-04	[illegible]
[illegible]	1.760D-05	0.99998240001548790	I	1	1.760D-04	[illegible]
[illegible]	1.024D-05	0.99998976000524288	I	1	[illegible]	11
[illegible]	1.024D-05	0.99998976000524288	I	1	[illegible]	11
[illegible]	1.725D-04	0.99982721492906600	I	1	1.500D-04	[illegible]
[illegible]	2.304D-05	0.99997696002654 1	I	1	1.500D-06	11
[illegible]	2.304D-05	0.99997696002654 1	I	1	1.500D-06	11
[illegible]	2.304D-05	0.99997696002654 1	I	1	1.500D-06	11
[illegible]	2.304D-05	0.99997696002654 1	I	1	1.500D-06	11
[illegible]	1.535D-03	0.99846517996448[illegible]	I	1	1.200D-03	11
[illegible]	1.535D-03	0.99846517996448[illegible]	I	1	1.200D-03	11
[illegible]	1.535D-03	0.99846517996448[illegible]	I	1	1.200D-03	11
[illegible]	2.201D-04	0.99977996[illegible]	I	1	1.720D-04	11
[illegible]	[illegible]D-03	0.99[illegible]0294	I	1	[illegible]D-03	11
[illegible]	1.349D-16	0.99999999999999[illegible]	I	1	[illegible]D-17	11
[illegible]	2.104D-04	0.99978962654004 16	I	1	1.200D-04	11
[illegible]	[illegible]	0.999999999[illegible]	[illegible]	4	[illegible]	[illegible]
[illegible]	1.327D-11	0.99999999999[illegible]	[illegible]	16	17	[illegible]
[illegible]	[illegible]D-12	0.99999999999976768	[illegible]	17	14	[illegible]
[illegible]	[illegible]D-12	0.99999999999976768	[illegible]	19	20	[illegible]
[illegible]	[illegible]D-16	0.9999999999999999[illegible]	[illegible]	1	[illegible]	
[illegible]	[illegible]D-16	0.9999999999999999[illegible]	[illegible]	1	[illegible]	
[illegible]	1.349D-16	0.99999999999999999[illegible]	0	[illegible]		
80	1.349D-16	0.99999999999999999[illegible]	0	[illegible]		
[illegible]	1.349D-16	0.99999999999999999[illegible]	0	[illegible]		
[illegible]	[illegible]	0.999999[illegible]4671	0	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	0.99994[illegible]14257[illegible]	0	[illegible]	[illegible]	[illegible]
84	4.[illegible]D-1[illegible]	0.99999999999[illegible]	0	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	0.999999[illegible]	[illegible]	11	[illegible]	
[illegible]	1.349D-16	0.99999999999999999	0	[illegible]	[illegible]	
[illegible]	4.607D-04	0.99953[illegible]061530144	0	[illegible]	10	
[illegible]	4.607D-04	0.99953[illegible]061530144	0	[illegible]	[illegible]	
[illegible]	1.327D-11	0.99999999999[illegible]	[illegible]	74	[illegible]	
90	[illegible]D-03	0.99[illegible]0294	[illegible]	[illegible]	[illegible]	[illegible]
91	[illegible]D-06	0.999999[illegible]2049512	[illegible]	[illegible]		
92	[illegible]D-06	0.99999[illegible]424671	[illegible]	[illegible]		
93	1.725D-04	0.99982721492[illegible]600	0	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	0.99[illegible]1452[illegible]	0	[illegible]		
[illegible]	[illegible]	0.99994[illegible]01425754[illegible]	[illegible]	[illegible]		
[illegible]	4.[illegible]D-1[illegible]	0.99999999999[illegible]	[illegible]	[illegible]		
[illegible]	[illegible]	0.99999[illegible]	[illegible]	[illegible]		
[illegible]	[illegible]	0.99[illegible]4545[illegible]	[illegible]	[illegible]	[illegible]	
[illegible]	1.[illegible]D-11	0.99999999999[illegible]	[illegible]	[illegible]		
100	4.711D-0[illegible]	0.[illegible]0477244	0	[illegible]	91	[illegible]

TERMINATIONS SPECIFIED ARE
[illegible]

[illegible] FAILURE COMBINATIONS, APPROXIMATION CUTOFFS: [illegible] [illegible]

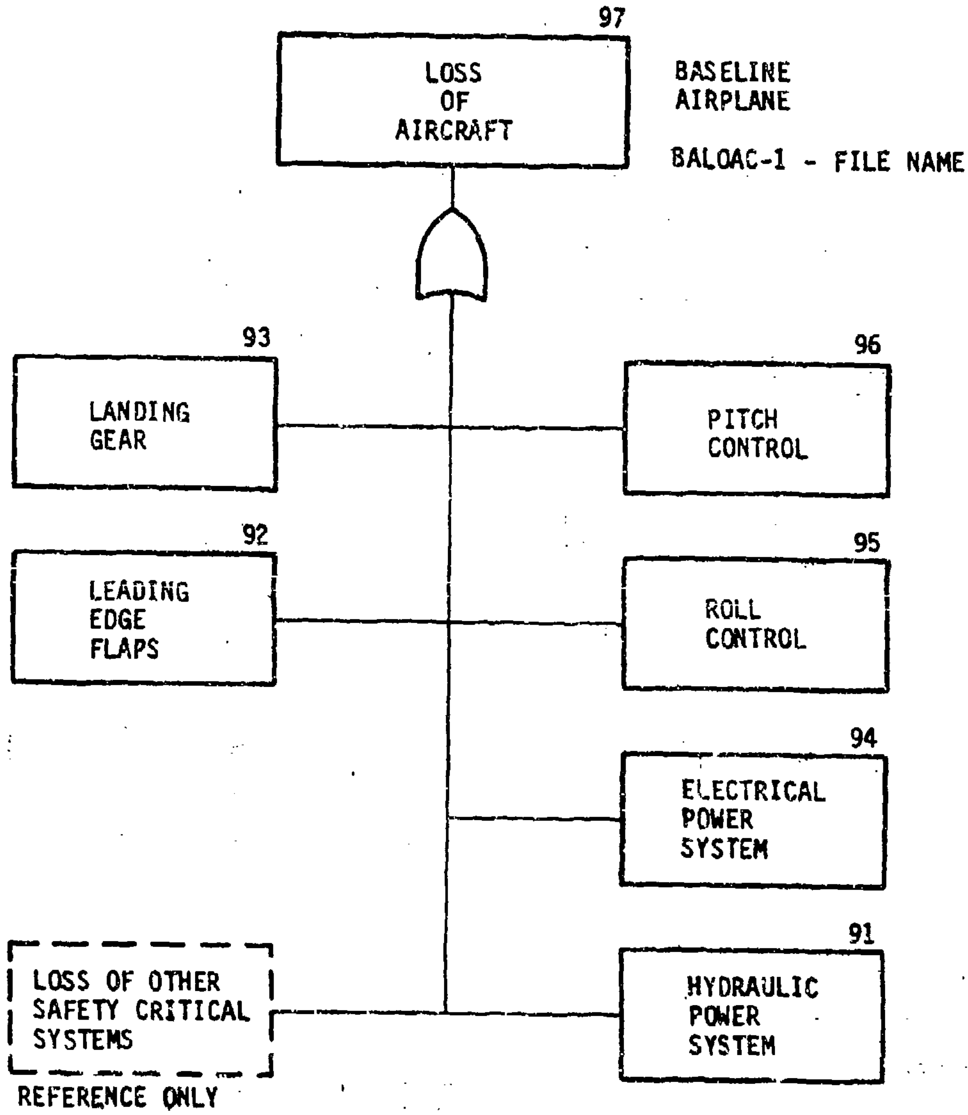


Figure A-24 Loss of Aircraft Fault Tree - Baseline Airplane

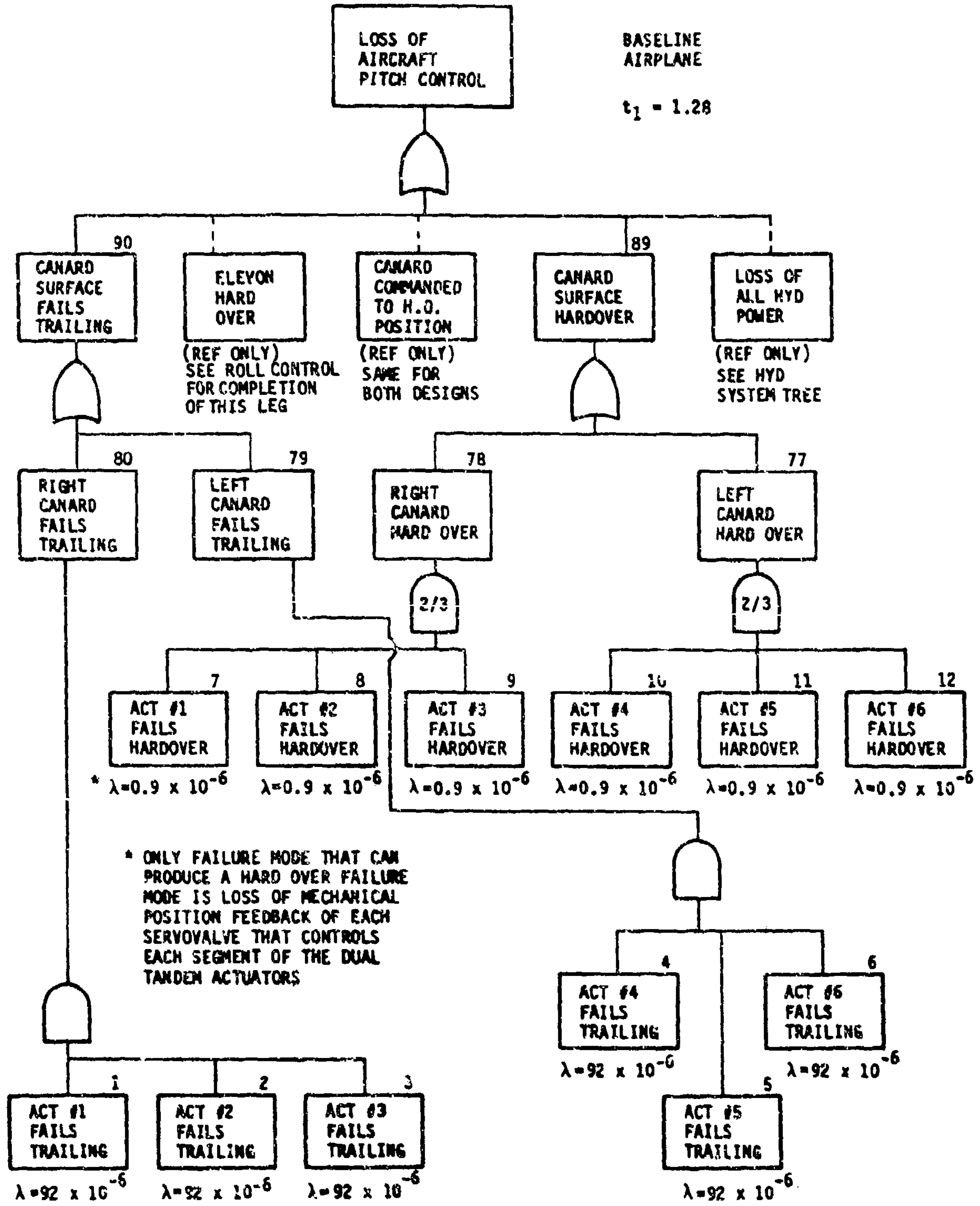


Figure A-25 Loss of Aircraft Fault Tree - Pitch Control Baseline Airplane

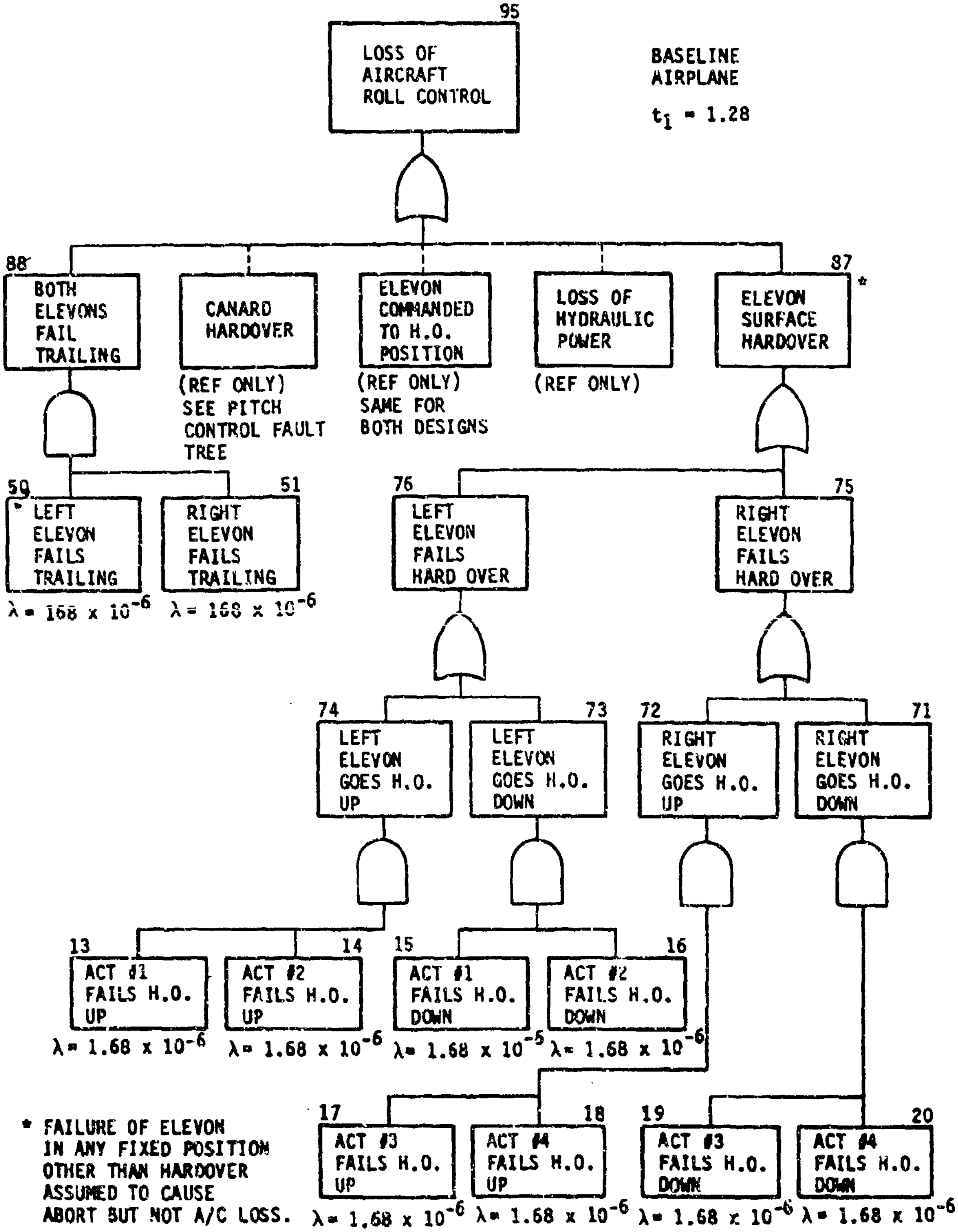


Figure A-26 Loss of Aircraft Fault Tree - Roll Control Baseline Airplane

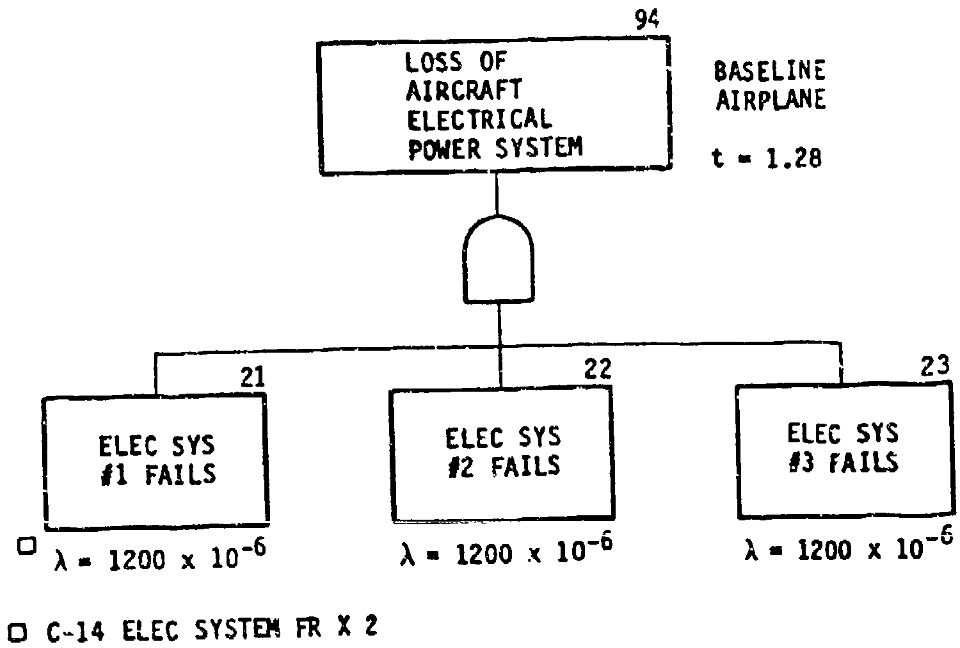
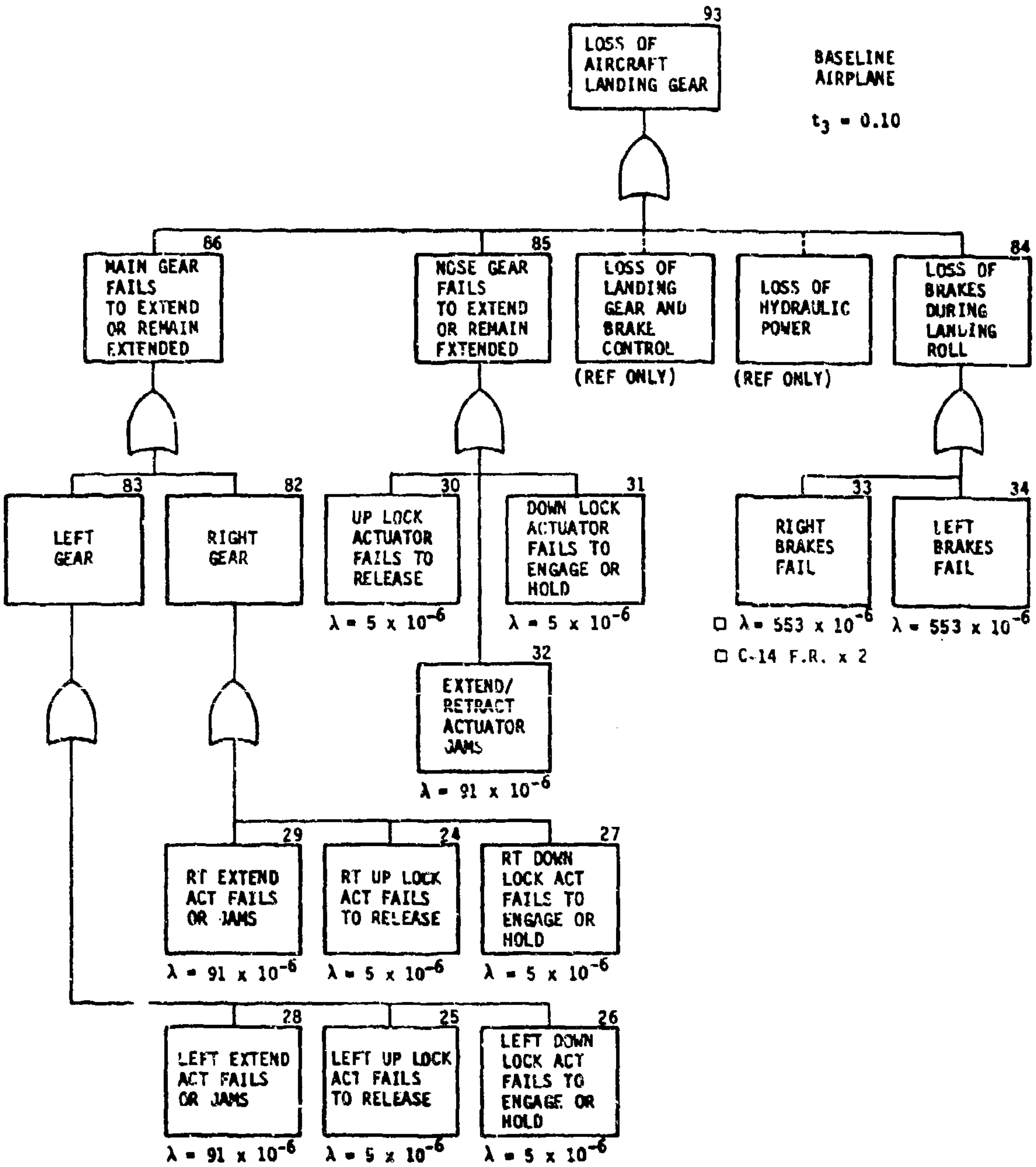
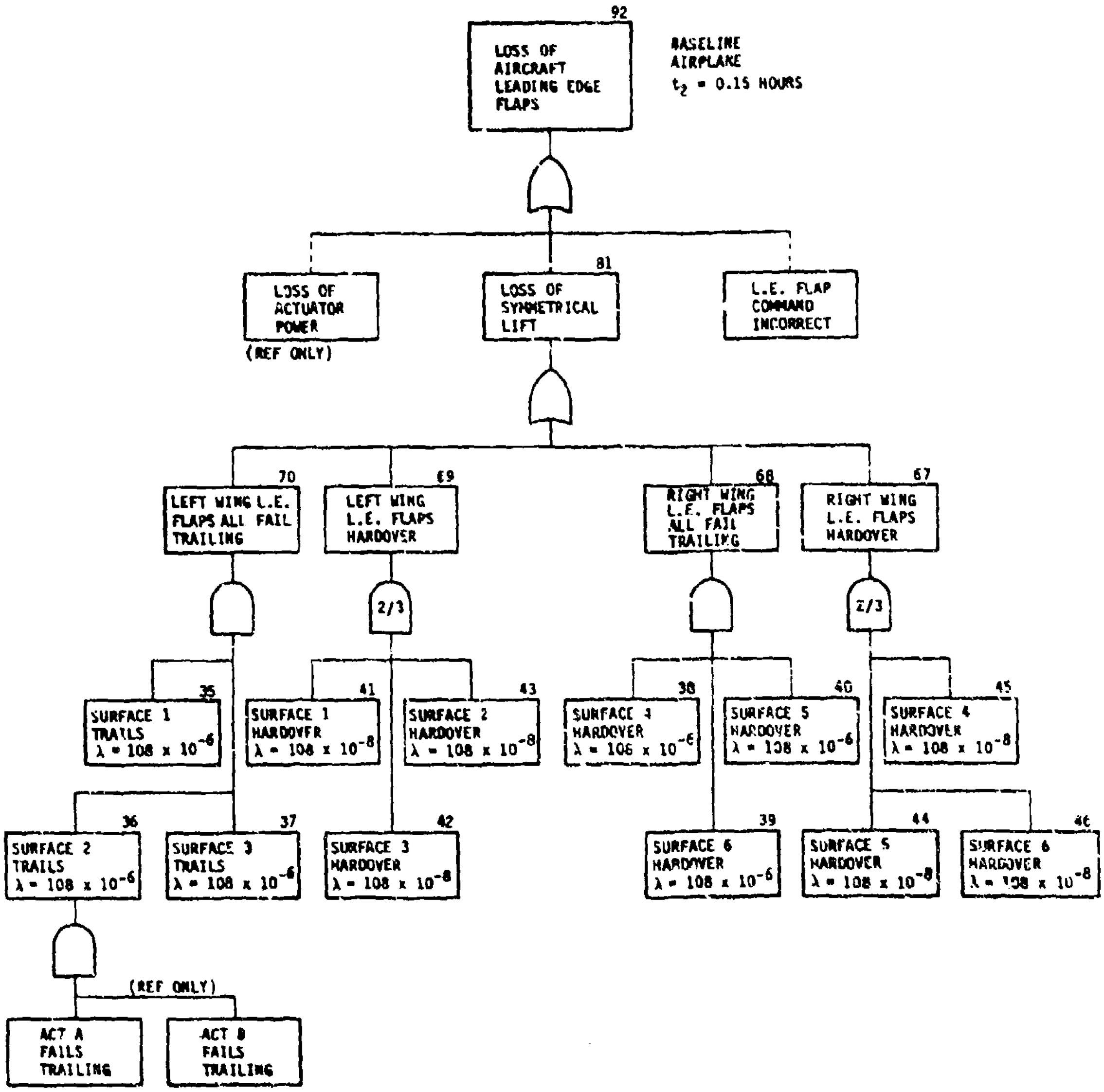


Figure A-27 Loss of Aircraft Fault Tree - Electrical Power System Baseline Airplane



- NOSE GEAR RETRACTS FORWARD. GEAR CAN BE EXTENDED BY FREE-FALL IF NOT JAMMED OR LOCKED.
- NOSE GEAR STEERING ASSUMED TO BE USED DURING TAXI ONLY AND IS NOT SAFETY CRITICAL.
- MAIN GEAR (PER SIDE) = 1 LINEAR PISTON ACT + 1 SOLENOID VALVE + POSIT INDICATOR $= 91 \times 10^{-6}$

Figure A-28 Loss of Aircraft Fault Tree - Landing Gear Baseline Airplane



- L.E. FLAPS REQUIRED FOR T.O. AND SOMETIMES USED FOR LANDING (EXPOSURE TIME 0.05 TAKEOFF AND 0.10 DURING LANDING)
- LOSS OF ALL FLAPS ON ONE WING DURING LANDING OR T/O CAUSES LOSS OF A/C

Figure A-29 Loss of Aircraft Fault Tree - Leading Edge Flaps Baseline Airplane

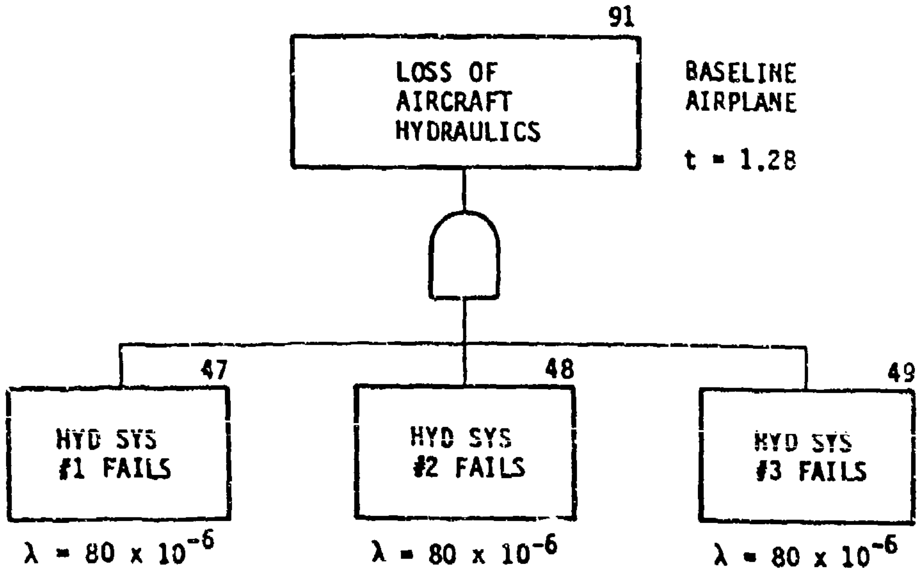


Figure A-30 Loss of Aircraft Fault Tree - Hydraulic Power System Baseline Airplane

TABLE A-3 AIRPLANE SAFETY - BASELINE AIRPLANE

(SHEET 1 OF 2)

BASELINE AIRPLANE - [illegible]

SECTION 3. SUMMARY OF THE SYSTEM PROBABILITIES.

FLIGHT HOURS (FH): 1= 1.150D+00 2= 1.500D-01
3= 1.000D-01

EVENT	UNRELIABILITY	RELIABILITY	GATE TYPE		NUMBER GATE / GATE INPUT	EV
1	1.173D-04	0.9998827449934366	I	1	9.100D-05	11
2	1.173D-04	0.9998827449934366	I	1	9.100D-05	11
3	1.173D-04	0.9998827449934366	I	1	9.100D-05	11
4	1.173D-04	0.9998827449934366	I	1	9.100D-05	11
5	1.173D-04	0.9998827449934366	I	1	9.100D-05	11
6	1.173D-04	0.9998827449934366	I	1	9.100D-05	11
7	1.550D-06	0.9999984500006634	I	1	9.000D-07	11
8	1.550D-06	0.9999984500006634	I	1	9.000D-07	11
9	1.550D-06	0.9999984500006634	I	1	9.000D-07	11
10	1.550D-06	0.9999984500006634	I	1	9.000D-07	11
11	1.550D-06	0.9999984500006634	I	1	9.000D-07	11
12	1.550D-06	0.9999984500006634	I	1	9.000D-07	11
13	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
14	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
15	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
16	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
17	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
18	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
19	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
20	2.150D-06	0.9999978496023121	I	1	1.650D-06	11
21	1.535D-03	0.9984651750442520	I	1	1.200D-03	11
22	1.535D-03	0.9984651750442520	I	1	1.200D-03	11
23	1.535D-03	0.9984651750442520	I	1	1.200D-03	11
24	5.000D-07	0.9999995000001249	I	1	5.000D-06	21
25	5.000D-07	0.9999995000001249	I	1	5.000D-06	21
26	5.000D-07	0.9999995000001249	I	1	5.000D-06	21
27	5.000D-07	0.9999995000001249	I	1	5.000D-06	21
28	9.100D-06	0.9999990900004140	I	1	9.100D-05	21
29	9.100D-06	0.9999990900004140	I	1	9.100D-05	21
30	5.000D-07	0.9999995000001249	I	1	5.000D-06	21
31	5.000D-07	0.9999995000001249	I	1	5.000D-06	21
32	9.100D-06	0.9999990900004140	I	1	9.100D-05	21
33	5.530D-05	0.9999447015290165	I	1	5.530D-04	21
34	5.530D-05	0.9999447015290165	I	1	5.530D-04	21
35	1.620D-05	0.9999838001312196	I	1	1.620D-04	21
36	1.620D-05	0.9999838001312196	I	1	1.620D-04	21
37	1.620D-05	0.9999838001312196	I	1	1.620D-04	21
38	1.620D-05	0.9999838001312196	I	1	1.620D-04	21
39	1.620D-05	0.9999838001312196	I	1	1.620D-04	21
40	1.620D-05	0.9999838001312196	I	1	1.620D-04	21
41	1.620D-07	0.9999998380000130	I	1	1.620D-06	21
42	1.620D-07	0.9999998380000130	I	1	1.620D-06	21
43	1.620D-07	0.9999998380000130	I	1	1.620D-06	21
44	1.620D-07	0.9999998380000130	I	1	1.620D-06	21
45	1.620D-07	0.9999998380000130	I	1	1.620D-06	21
46	1.620D-07	0.9999998380000130	I	1	1.620D-06	21
47	1.024D-04	0.9998976052427010	I	1	9.000D-05	11
48	1.024D-04	0.9998976052427010	I	1	9.000D-05	11
49	1.024D-04	0.9998976052427010	I	1	9.000D-05	11
50	2.150D-04	0.9997849902119436	I	1	1.650D-04	11
51	2.150D-04	0.9997849902119436	I	1	1.650D-04	11
67	7.373D-14	0.9999999999999262	O	44	45	46
68	3.334D-15	0.9999999999999971	O	38	39	40
69	7.373D-14	0.9999999999999262	O	41	42	43
70	4.351D-15	0.9999999999999956	A	35	36	
71	4.624D-13	0.9999999999995377	A	19	20	
72	4.624D-13	0.9999999999995377	A	17	18	
73	4.624D-13	0.9999999999995377	A	15	16	
74	4.624D-13	0.9999999999995377	A	13	14	
75	9.245D-13	0.9999999999990755	O	21	22	
76	9.245D-13	0.9999999999990755	O	23	24	

TABLE A-3 AIRPLANE SAFETY - BASELINE AIRPLANE (SHEET 2 OF 2)

[illegible]	[illegible]	0.[illegible]0187	[illegible]	10	11	[illegible]		
[illegible]	[illegible]	0.[illegible]0187	[illegible]	[illegible]	[illegible]			
[illegible]	[illegible]	0.[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		
[illegible]	[illegible]	0.[illegible]	[illegible]	1	[illegible]			
[illegible]	[illegible]	0.[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	
[illegible]	1.010D-05	0.[illegible]10047	[illegible]	[illegible]	[illegible]	[illegible]		
[illegible]	[illegible]	0.[illegible]	[illegible]	[illegible]	[illegible]			
84	1.106D-04	0.[illegible]11[illegible]9543	0	[illegible]	[illegible]			
85	1.010D-05	0.[illegible]510047	0	50	51	[illegible]		
86	1.110D-05	0.[illegible]16047	0	[illegible]	[illegible]			
87	1.850D-11	0.[illegible]15039	0	[illegible]	[illegible]			
88	4.623D-08	0.[illegible]37677411	4	50	51			
89	7.963D-12	0.[illegible]20374	0	[illegible]	[illegible]			
90	3.265D-12	0.[illegible]67345	0	79	[illegible]			
91	1.074D-12	0.[illegible]263	6	47	[illegible]	[illegible]		
92	1.645D-11	0.[illegible]6353	0	51				
93	[illegible]	0.[illegible]52384	0	94	[illegible]	[illegible]		
94	[illegible]	0.[illegible]544600	4	21	[illegible]	[illegible]		
95	[illegible]	0.[illegible]7492442	0	[illegible]	[illegible]			
96	1.133D-11	0.[illegible]65771[illegible]	0	66	[illegible]			
97	1.319D-04	0.[illegible]15881[illegible]501	0	91 [illegible]	[illegible]	[illegible]	94	[illegible]

[illegible]ENCIES SPECIFIED ARE

* FAILURE COMBINATIONS, APPROXIMATION CUTOFF: [illegible]

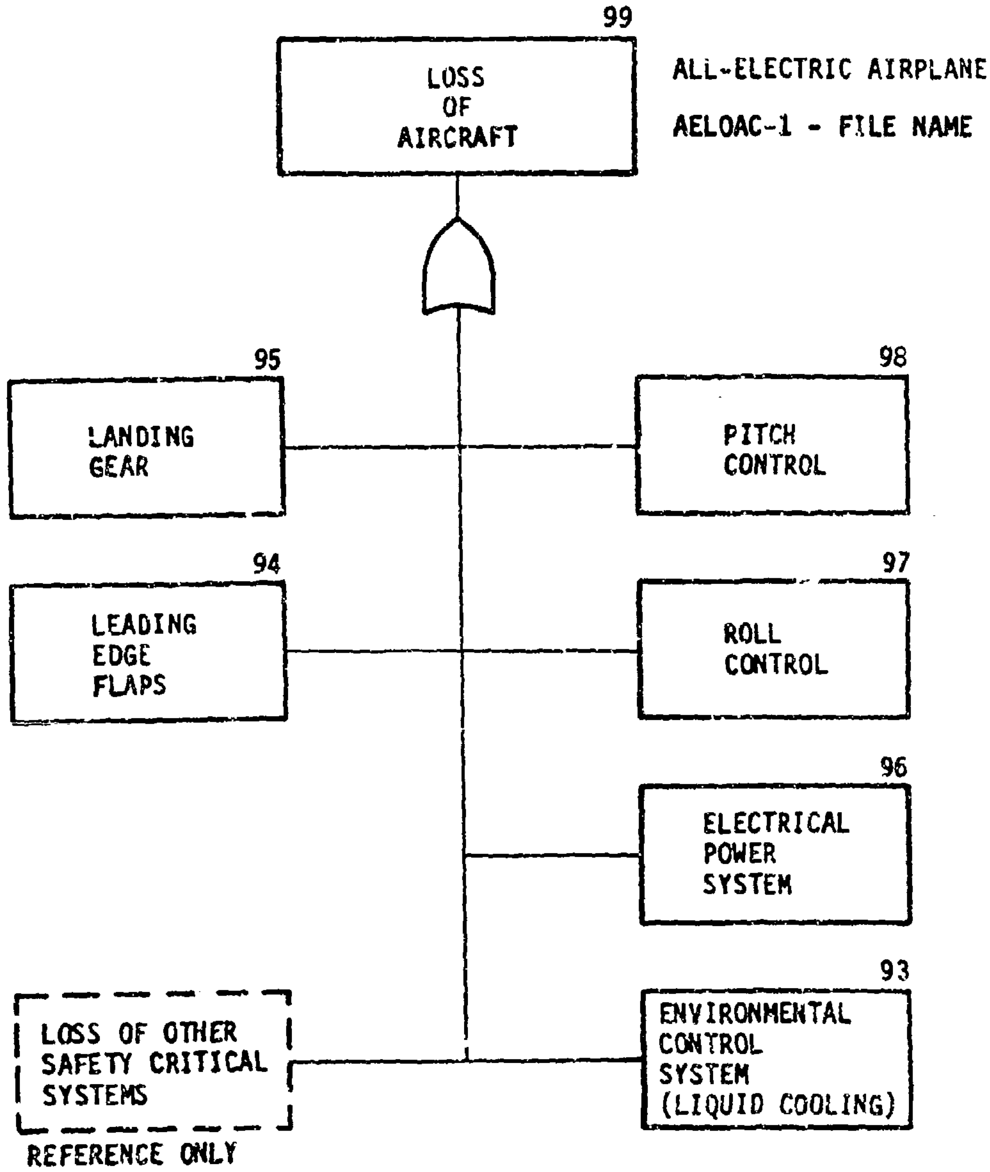


Figure A-31 Loss of Aircraft Fault Tree - All-Electric Airplane

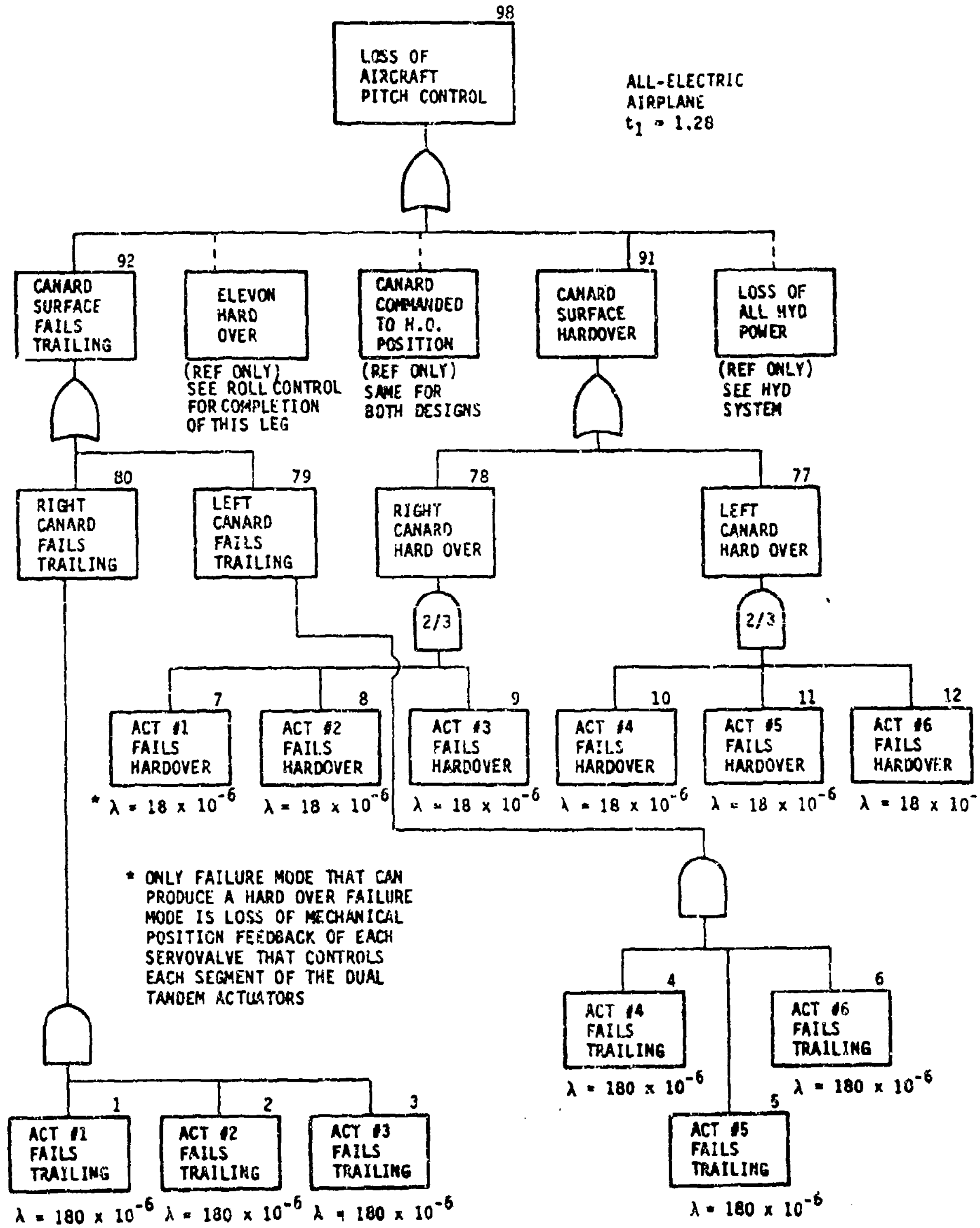


Figure A-32 Loss of Aircraft Fault Tree - Pitch Control All-Electric Airplane

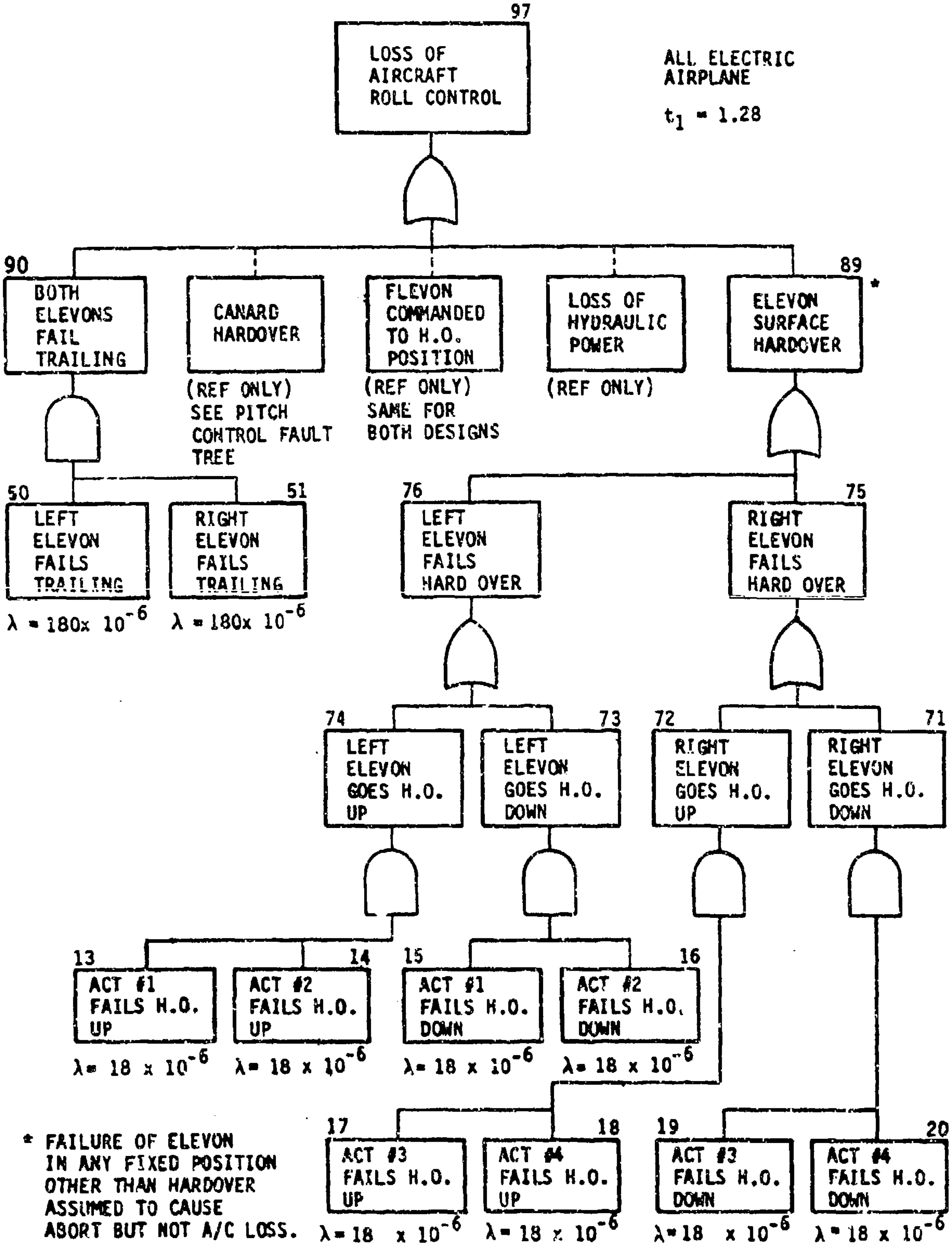


Figure A-33 Loss of Aircraft Fault Tree - Roll Control All-Electric Airplane

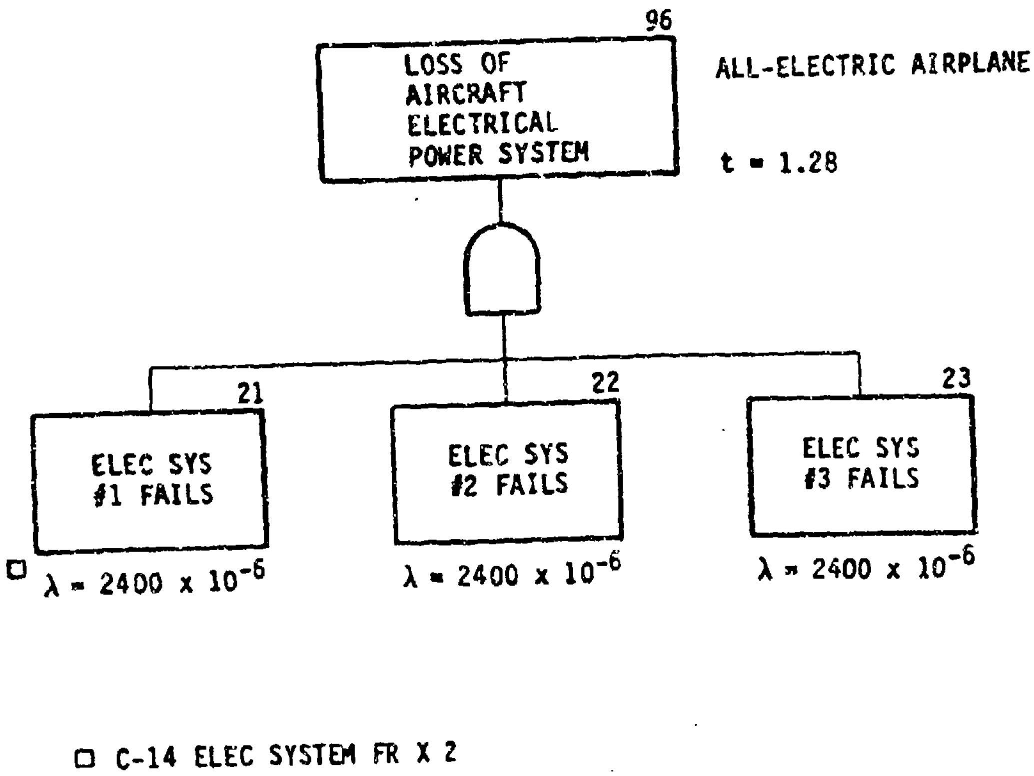
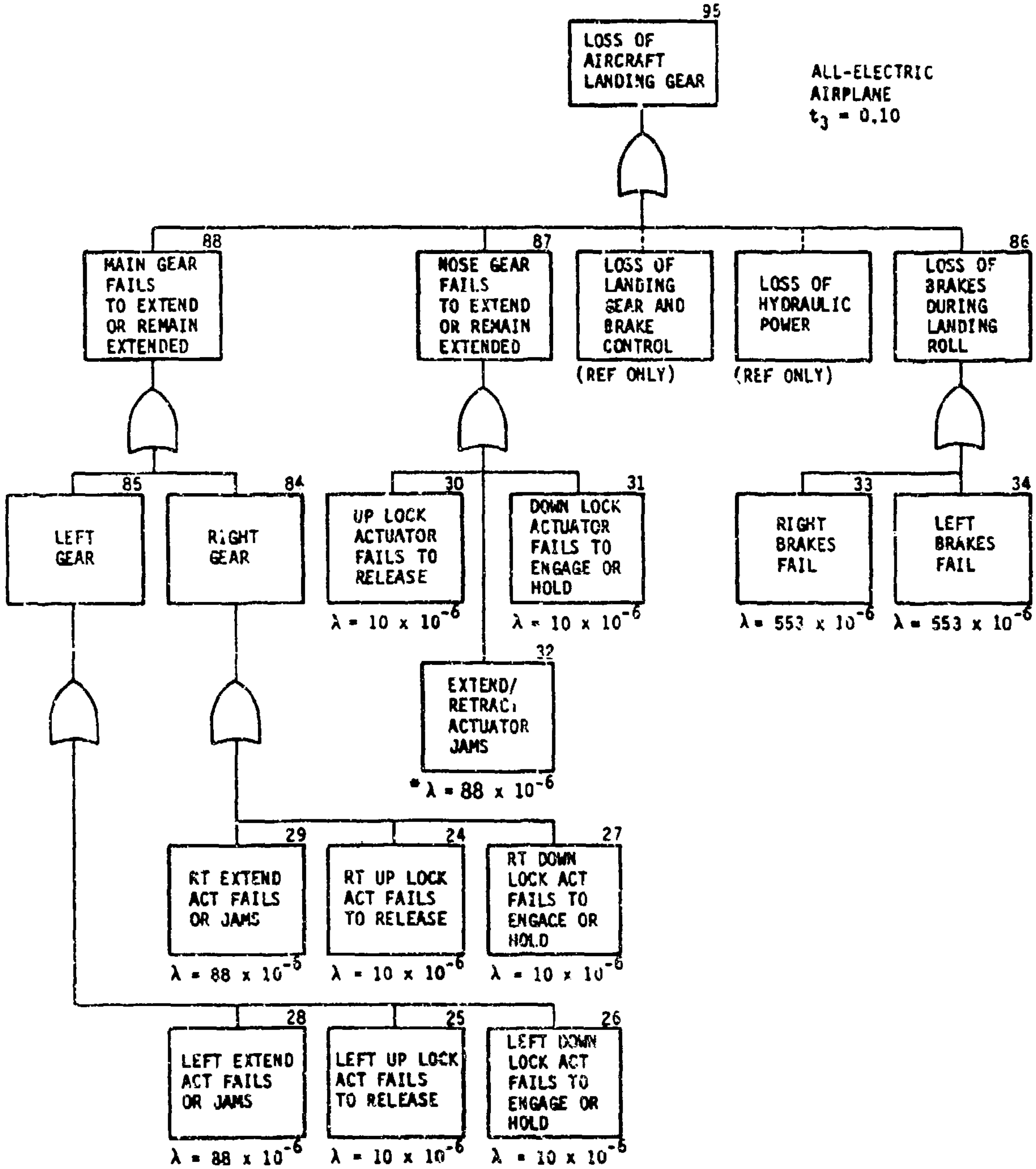


Figure A-34 Loss of Aircraft Fault Tree - Electrical Power System All-Electric Airplane

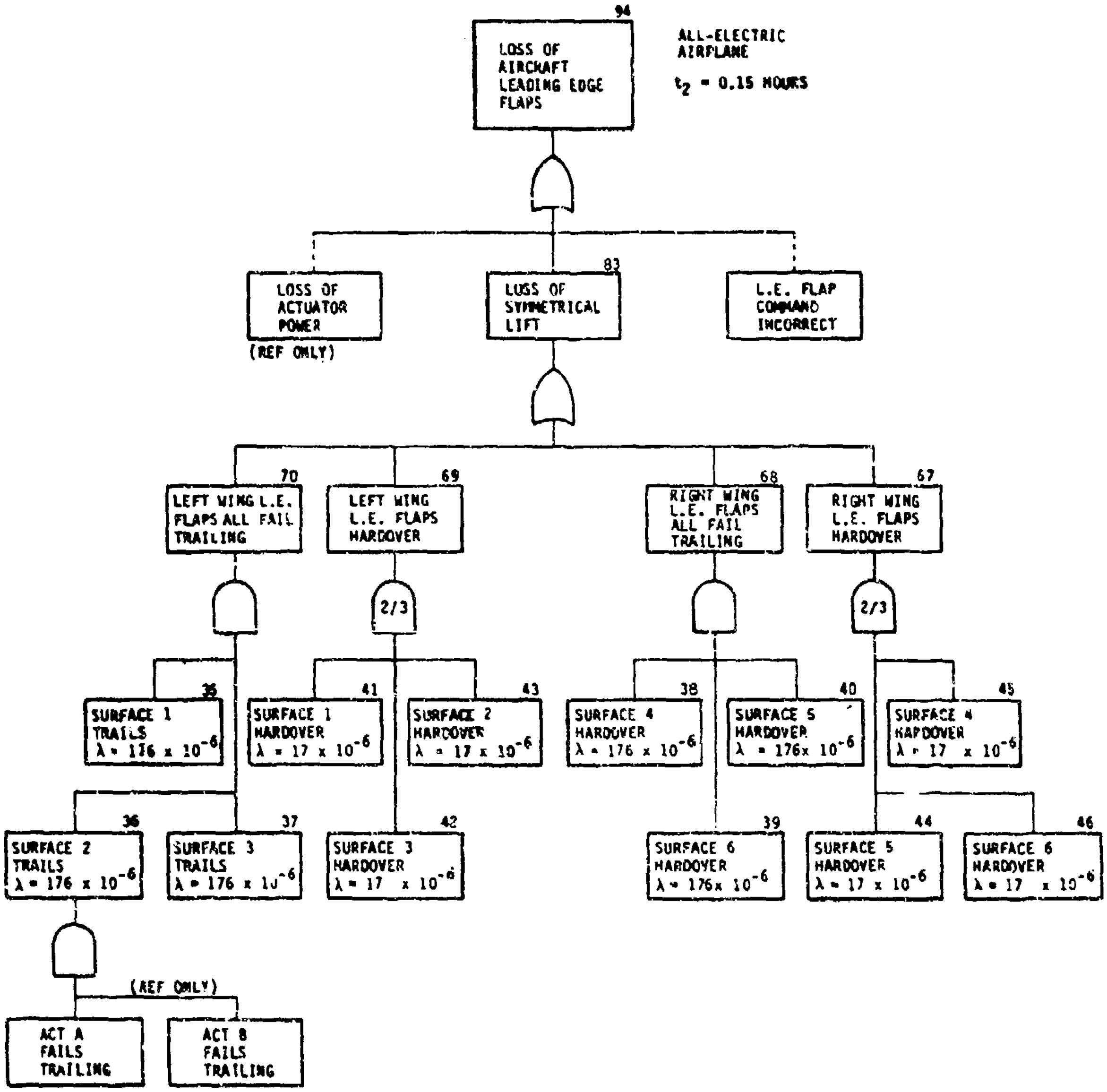


• NOSE GEAR RETRACTS FORWARD. GEAR CAN BE EXTENDED BY FREE-FALL IF NOT JAMMED OR LOCKED.

• NOSE GEAR STEERING ASSUMED TO BE USED DURING TAXI ONLY AND IS NOT SAFETY CRITICAL.

* USE 1/2 OF L.E. FLAP ACT = (1/2)(176) = 88×10^{-6}

Figure A-35 Loss of Aircraft Fault Tree - Landing Gear All-Electric Airplane



- L.E. FLAPS REQUIRED FOR T.O. AND SOMETIMES USED FOR LANDING (EXPOSURE TIME 0.05 TAKEOFF AND 0.10 DURING LANDING)
- LOSS OF ALL FLAPS ON ONE WING DURING LANDING OR T/O CAUSES LOSS OF A/C

Figure A-36 Loss of Aircraft Fault Tree - Leading Edge Flaps All-Electric Airplane

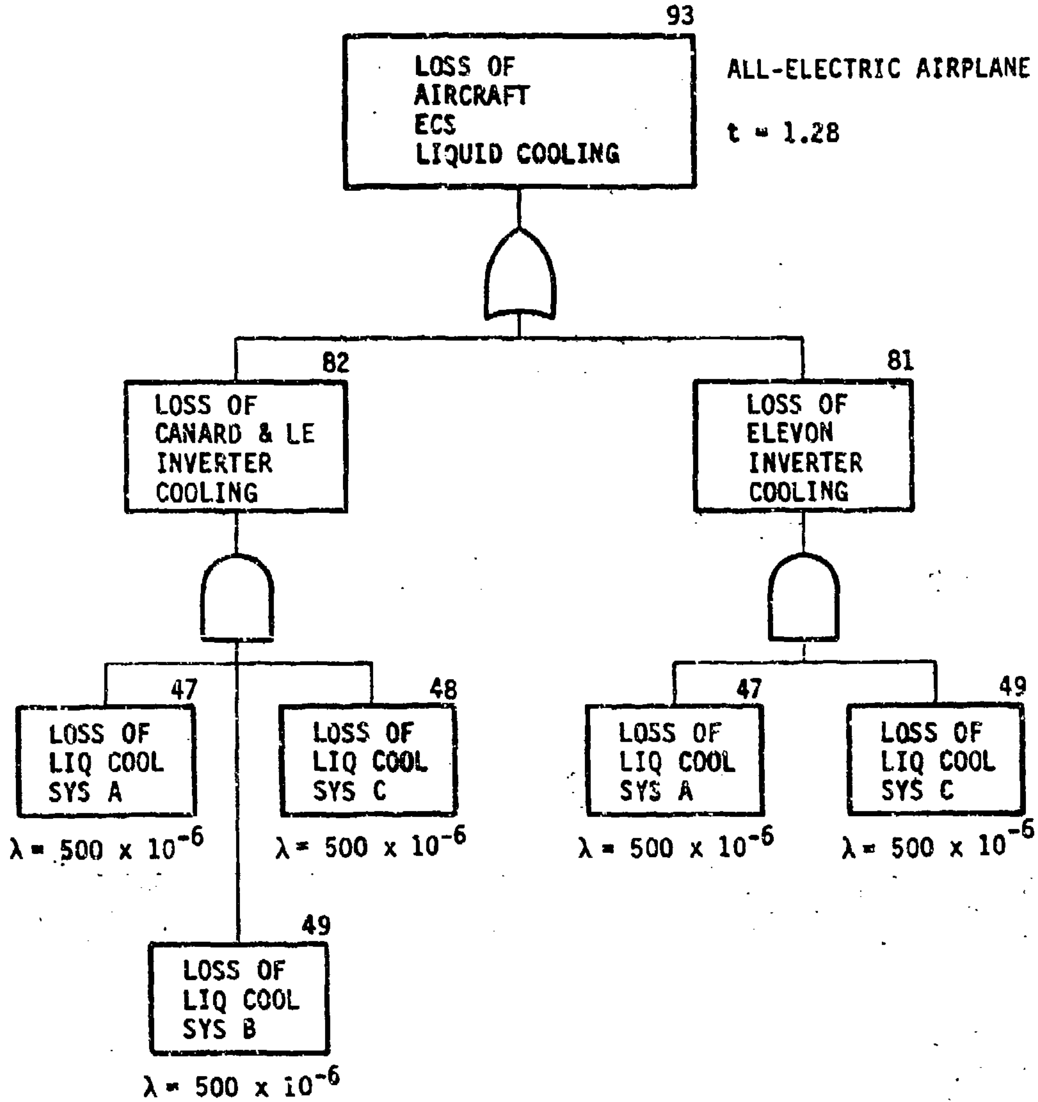


Figure A-37 Loss of Aircraft Fault Tree - ECS Liquid Cooling System All-Electric Airplane

TABLE A-4 AIRPLANE SAFETY - ALL-ELECTRIC AIRPLANE
(SHEET 1 OF 2)

ALL-ELECTRIC AIRPLANE - LOSS OF A/C - [illegible]

SECTION 3. SUMMARY OF THE SYSTEM PROBABILITIES.

FLIGHT HOURS (FH): T= 1.590D+00 S= 1.500D-01
 = 1.000D-01

EVENT	UNRELIABILITY	RELIABILITY	GATE TYPE		FAILURE RATE OR GATE INPUT		FAI...
1	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
2	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
3	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
4	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
5	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
6	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
7	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
8	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
9	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
10	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
11	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
12	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
13	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
14	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
15	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
16	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
17	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
18	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
19	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
20	2.304D-05	0.99997696026541 59	I	1	1.500D-05		11
21	3.067D-03	0.99693271376 35703	I	1	2.400D-03		11
22	3.067D-03	0.99693271376 35703	I	1	2.400D-03		11
23	3.067D-03	0.99693271376 35703	I	1	2.400D-03		11
24	1.000D-06	0.99999900000004999	I	1	1.000D-06		21
25	1.000D-06	0.99999900000004999	I	1	1.000D-06		21
26	1.000D-06	0.99999900000004999	I	1	1.000D-06		21
27	1.000D-06	0.99999900000004999	I	1	1.000D-06		21
28	5.000D-06	0.99999500001 25 7199	I	1	5.000D-06		21
29	5.000D-06	0.99999500001 25 7199	I	1	5.000D-06		21
30	1.000D-06	0.99999900000004999	I	1	1.000D-06		21
31	1.000D-06	0.99999900000004999	I	1	1.000D-06		21
32	1.000D-06	0.99999900000004999	I	1	1.000D-06		21
33	5.530D-05	0.99994470152901 65	I	1	5.530D-04		21
34	5.530D-05	0.99994470152901 65	I	1	5.530D-04		21
35	2.640D-05	0.99997360034847 69	I	1	1.760D-04		21
36	2.640D-05	0.99997360034847 69	I	1	1.760D-04		21
37	2.640D-05	0.99997360034847 69	I	1	1.760D-04		21
38	2.640D-05	0.99997360034847 69	I	1	1.760D-04		21
39	2.640D-05	0.99997360034847 69	I	1	1.760D-04		21
40	2.640D-05	0.99997360034847 69	I	1	1.760D-04		21
41	2.550D-06	0.99999745000325 12	I	1	1.700D-05		21
42	2.550D-06	0.99999745000325 12	I	1	1.700D-05		21
43	2.550D-06	0.99999745000325 12	I	1	1.700D-05		21
44	2.550D-06	0.99999745000325 12	I	1	1.700D-05		21
45	2.550D-06	0.99999745000325 12	I	1	1.700D-05		21
46	2.550D-06	0.99999745000325 12	I	1	1.700D-05		21
47	5.200D-04	0.99948020475 63163	I	1	5.000D-04		11
48	5.200D-04	0.99948020475 63163	I	1	5.000D-04		11
49	5.200D-04	0.99948020475 63163	I	1	5.000D-04		11
50	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
51	2.304D-04	0.99976962654004 16	I	1	1.500D-04		11
67	1.951D-11	0.99999999998049 25	2	44	45	46	
68	1.540D-14	0.99999999999999 16	6	36	39	40	
69	1.951D-11	0.99999999998049 25	2	41	42	43	
70	1.840D-14	0.99999999999999 16	6	35	36	37	
71	5.306D-10	0.99999999946941705	6	19	20		
72	5.306D-10	0.99999999946941705	6	17	18		
73	5.306D-10	0.99999999946941705	6	15	16		
74	5.306D-10	0.99999999946941705	6	13	14		
75	1.062D-09	0.99999999893 3412	0	73	71		
76	1.062D-09	0.99999999893 3412	0	72	74		
77	1.552D-09	0.99999999844 07 [illegible]	3	10	11	12	
78	1.552D-09	0.99999999844 07 [illegible]	3	7	[illegible]	[illegible]	
79	1.221D-11	0.99999999998 7776	6	4	[illegible]	[illegible]	
80	1.221D-11	0.99999999998 7776	6	1	[illegible]	[illegible]	

TABLE A-4 AIRPLANE SAFETY - ALL-ELECTRIC AIRPLANE

(SHEET 2 OF 2)

81	4.093D-07	0.9999995906620461	0	47	49				
82	3.619D-10	0.9999999996739107	0	47	48	49			
83	7.905D-11	0.9999999999209454	0	67	69	69	70		
84	1.090D-05	0.9999890000593199	0	64	67	69			
85	1.090D-05	0.9999890000593199	0	65	66	69			
86	1.106D-04	0.9998894061159545	0	73	74				
87	3.000D-06	0.9999970000045000	0	70	71	73			
88	2.160D-05	0.9999784092232783	0	94	94				
89	2.127D-09	0.9999999978766524	0	75	76				
90	5.307D-06	0.9999946928686629	0	79	81				
91	3.159D-06	0.9999968150736	0	77	78				
92	2.445D-11	0.9999999999755473	0	79	80				
93	4.093D-07	0.9999995906620461	0	81	82				
94	7.905D-11	0.9999999999209454	0	83					
95	1.353D-04	0.9998648001391988	0	86	87	88			
96	2.556D-09	0.9999999971142229	0	21	82	83			
97	5.520D-06	0.9999944804756715	0	89	90				
98	3.209D-09	0.9999999967906199	0	91	92				
99	1.357D-04	0.9998643125668728	0	93	94	95	96	97	
				98					

DEPENDENCIES SPECIFIED ARE
47 49

2 FAILURE COMBINATIONS. APPROXIMATION CUTOFFS: 1.00D-10 1.56D-15

APPENDIX B

RCA PRICE-L COST MODEL INPUT DATA

This appendix contains the RCA PRICE-L cost model input data for 500 Baseline and All-Electric Airplanes. Tables B-1 and B-2 contain data for the actuation systems and Tables B-3 and B-4 are for the secondary power systems. The input data for 1000 airplanes is identical except for the "Production Quantity (QTY)" number on the input data sheet (see Figure 35). For 500 airplanes, QTY is 500 times QTYNHA; for 1000 airplanes QTY is 1000 times QTYNHA.

TABLE B-1 RCA PRICE MODEL INPUT DATA-BASELINE AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 1 OF 3

```
J HIHY1.DAT

HIHY1.DAT  25-SEP-81  11:38

**PRICE 84
LINEAR ACTUATOR-CANARD          MODIFIED 8-3-81
2000 60 39 .1625 2
4 0 .5 1.8 1981
39 5.8 .1 0 0 .33
190 C C 1
195 C C 55
HYDRAULIC SERVO VALVE - CANARD
1090 30 7 .0292 1
2 .5 .3 1.8 1981
4.45 5.8 .1 0 0 .33
40 7.940 .1 0 0 1.0
0190 C C 1
0195 C C 10055
LINEAR ACTUATOR-ELEVON
2000 60 75 .31 1
4 .5 .5 1.8 1981
74.5 5.8 .1 0 0 .33
40 7.940 .1 0 0 1
0190 C C 1
0195 C C 10055
ROTARY MOTOR - RUDDER
1000 30 7.5 .0313 1
2 .5 .3 1.8 1981
7.13 5.80 .1 0 0 .33
40 7.940 .1 0 0 1
0190 C C 1
0195 C C 10055
HINGELINE GEAR BOX - RUDDER
500 15 22 .0957 2
1 0 .5 1.8 1981
22 5.8 .1 0 0 .33
0190 C C 1
0195 C C 55
REDUCTION GEAR BOX - RUDDER
500 15 11 .0478 2
1 0 .5 1.8 1981
11 5.8 .1 0 0 .33
0190 C C 1
0195 C C 55
LINEAR ACTUATOR - SPOILER
2000 60 17.8 .0937 1
4 .5 .5 1.8 1981
17.3 5.8 .1 0 0 .33
40 7.94 .1 0 0 1
190 C C 1
195 C C 10055
LINEAR ACTUATOR - LE FLAP
6000 180 19.3 .1016 1
12 .5 .5 1.8 1981
18.8 5.8 .1 0 0 .33
40 7.94 .1 0 0 1
190 C C 1
195 C C 10055
LINEAR ACTUATOR - ENGINE INLET CENTERBODY
1000 30 18 .0947 1
2 .5 .5 1.8 1981
17.5 5.8 .1 0 0 .33
40 7.94 .1 0 0 1
190 C C 1
195 C C 10055
ROTARY GEAR BOX - ENGINE INLET BYPASS DOOR
2000 60 2 .0087 1
4 .5 .5 1.8 1981
1.9 5.8 .1 0 0 .33
40 7.94 .1 0 0 1
190 C C 1
195 C C 10055
```

TABLE B-1 RCA PRICE MODEL INPUT DATA-BASELINE AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 2 OF 3

```
00770   HYDRAULIC MOTOR - ENGINE INLET BYPASS DOOR
00780   2000 40 3 .0083 2
00790   4 0 .3 1.8 1981
00800   2 5.8 .1 0 0 .33
00810   190 C C 1
00820   195 C C 55
00830   LINEAR ACTUATOR - MAIN LANDING GEAR
00840   1000 30 18.9 .0995 2
00850   2 0 .5 1.8 1981
00860   18.9 5.8 .1 0 0 .33
00870   190 C C 1
00880   195 C C 55
00890   LINEAR ACTUATOR - NOSE GEAR
00900   500 15 29.5 .1553 2
00910   1 0 .5 1.8 1981
00920   29.5 5.8 .1 0 0 .33
00930   190 C C 1
00940   195 C C 55
00950   CONTROL VALVE, 3 POSITION - LANDING GEAR
00960   500 15 3 .0125 1
00970   1 .5 .3 1.8 1981
00980   2.85 5.8 .1 0 0 .33
00990   40 7.94 .1 0 0 1
01000   190 C C 1
01010   195 C C 10055
01020   ACTUATOR - NOSE STEERING GEAR
01030   500 15 22 .0957 2
01040   1 0 .5 1.8 1981
01050   22 5.8 .1 0 0 .33
01060   190 C C 1
01070   195 C C 55
01080   ISOLATION VALVE - GROUND SYSTEM
01090   1000 30 2 .0083 1
01100   2 .5 .3 1.8 1981
01110   1.9 5.8 .1 0 0 .33
01120   40 7.94 .1 0 0 1
01130   190 C C 1
01140   195 C C 10055
01150   ACTUATOR - MAIN GEAR BRAKES
01160   1000 30 12 .0432 2
01170   2 0 .5 1.8 1981
01180   12 5.8 .1 0 0 .33
01190   190 C C 1
01200   195 C C 55
01210   CONTROL VALVE - MAIN GEAR BRAKES
01220   1000 30 9 .04 2
01230   2 0 .3 1.8 1981
01240   9 6.32 .1 0 0 .33
01250   190 C C 1
01260   195 C C 55
01270   SHUTOFF VALVE, MAIN GEAR BRAKES
01280   1000 30 1 .004 2
01290   2 0 .3 1.8 1991
01300   1 5.8 .1 0 0 .33
01310   190 C C 1
01320   195 C C 55
01330   PARKING VALVE - MAIN GEAR BRAKES
01340   1000 30 2.5 .0104 2
01350   2 0 .3 1.8 1981
01360   2.5 5.8 .1 0 0 .33
01370   190 C C 1
01380   195 C C 55
01390   ACCUMULATOR - MAIN GEAR BRAKES
01400   1000 30 10 .0781 2
01410   2 0 .3 1.8 1981
01420   10 5.78 .1 0 0 .33
01430   190 C C 1
01440   195 C C 55
01510   ACTUATOR-AERIAL REFUELING
01520   500 15 1.5 .0079 2
01530   1 0 .5 1.8 1981
01540   1.5 5.8 .1 0 0 .33
01550   0190 C C 1
01560   0195 C C 55
01570   ACTUATOR- AERIAL REFUELING NOZZLE LATCH
01580   500 15 1.0 .0053 2
01590   1 0 .5 1.8 1981
01600   1.0 5.8 .1 0 0 .33
01610   0190 C C 1
01620   0195 C C 55
```

TABLE B-1 RCA PRICE MODEL INPUT DATA-BASELINE AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 3 OF 3

```
01630   CONTROL VALVE- AERIAL REFUELING
01640   500 15 3.25 .0135 2
01650   1 0 .5 1.8 1981
01660   3.25 5.8 .1 0 0 .33
01670   0190 C C 1
01680   0195 C C 55
01690   LINEAR ACTUATOR- CANOPY
01700   500 15 2.9 .0153 2
01710   1 0 .5 1.8 1981
01720   2.9 5.8 .1 0 0 .33
01730   0190 C C 1
01740   0195 C C 55
01750   CONTROL VALVE. 3 POSITION- CANOPY
01760   500 15 1.0 .0042 2
01770   1 .5 .3 1.8 1981
01780   1.0 5.8 .1 0 0 .33
01790   0190 C C 1
01800   0195 C C 55
01810   GEAR BOX- GUN DRIVE
01820   500 15 10 .0435 2
01830   1 0 .5 1.8 1981
01840   10 5.8 .1 0 0 .33
01850   0190 C C 1
01860   0195 C C 55
01870   HYDRAULIC MOTOR -  GUN DRIVE
01880   500 15 7.6 .0317 2
01890   1 0 .3 1.8 1981
01900   7.6 5.8 .1 0 0 .33
01910   0190 C C 1
01920   0195 C C 55
01930   CONTROL VALVE , 3 POSITION - GUN DRIVE
01940   500 15 8.4 .0350 2
01950   1 0 .3 1.8 1981
01960   8.4 5.8 .1 0 0 .33
01970   0190 C C 1
01980   0195 C C 55
01990   HYDRAULIC MOTOR- ECS PACK COMPRESSOR
02000   500 15 4.0 .018 2
02010   1 0 .3 1.8 1981
02020   4.0 5.8 .1 0 0 .33
02030   0190 C C 1
02040   0195 C C 55
02050   CONTROL VALVE, 2 POSITION- ECS PACK COMPRESSOR
02055   500 15 1 .0042 1
02060   1 .5 .3 1.8 1981
02070   .95 5.8 .1 0 0 .33
02080   [illegible] 7.940 .1 0 0 1.0
02090   0190 C C 1
02100   0195 C C 10055
02110   GEAR BOX  - ELECTRONICS COOLING FAN
02120   500 15 7.5 .0326 2
02130   1 0 .5 1.8 1981
02140   7.5 5.8 .1 0 0 .33
02150   0190 C C 1
02160   0195 C C 55
02170   HYDRAULIC MOTOR - ELECTRONICS COOLING FAN
02180   500 15 7.6 .0317 2
02190   1 0 .3 1.9 1981
02200   7.6 5.8 .1   0 .33
02210   0190 C C 1
02220   0195 C C 55
02230   CONTROL VALVE, 3 POSITION- ELECTRONICS COOLING FAN
02240   500 15 1.0 .0042 1
02250   1 .5 .3 1.8 1981
02260   .95 5.8 .1 0 0 .33
02275   40 7.940 .1 0 0 1.0
02280   0190 C C 1
02290   0195 C C 10055
02300   SERVO VALVE - GENERAL PURPOSE
02310   12000 360 1.  .0042 1
02320   24 .5 .3 1.8 1981
02330   .95 5.8 .1 0 0 .33
02340   40 7.940 .1 0 0 1.0
02350   0190 C C 1
02360   0195 C C 10055
02370   INTEGRATION AND TEST
02380   500 15 .7 .7 5
02390   0 0 0 1.8 1981
02400   0190 C C 0195 C
```

TABLE B-2 RCA PRICE MODEL INPUT DATA-ALL-ELECTRIC AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 1 OF 5

```
HIEM1.DAT  23-SEP-81  16:39:12

**PRICE 84
POWER DRIVE UNIT/BALL SCREW ACT-CANARD      MODIFIED 8/3/81
1000 30 38.0 .1652 2
2 0 .5 1.8 1981
38 5.8 .1 0 0 .33
0190 C C 1
0195 C C 55
MOTOR - CANARD
3000 90 8 .0421 2
6 0 .3 1.8 1981
8 5.3 .1 0 0 .33
0190 C C 1
0195 C C 55
INVERTER - CANARD
3000 90 7.7 .07 1
6 .3 .3 1.8 1981
6 5.52 .1 0 0 .33
51 6.941 .1 0 0 1
0190 C C 1.0
0195 C C 10453
POWER DRIVE UNIT / HINGELINE GEAR BOX - ELEVON
1000 30 70 .3043 2
2 0 .5 1.8 1981
70 5.8 .1 0 0 .33
0190 C C 1
0195 C C 55
MOTOR - ELEVON
2000 60 13.7 .0628 2
4 0 .3 1.8 1981
13.7 5.3 .1 0 0 .33
0190 C C 1
0195 C C 55
INVERTER- ELEVON
3000 60 24.5 .2227 1
4 .3 .3 1.8 1981
22.1 5.52 .1 0 0 .33
51 6.941 .1 0 0 1
0190 C C 1.0
0195 C C 10453
POWER DRIVE UNIT / HINGELINE GEAR BOX  - RUDDER
500 15 39 .1696 2
1 0 .5 1.8 1981
39 5.8 .1 0 0 .33
0190 C C 1
0195 C C 55
MOTOR - RUDDER
1000 30 10.5 .0517 2
2 0 .3 1.8 1981
10.5 5.3 .1 0 0 .33
0190 C C 1
0195 C C 55
INVERTER - RUDDER
1000 30 14 .1273 1
2 .3 .3 1.8 1981
12.4 5.52 .1 0 0 .33
51 6.941 .1 0 0 1
0190 C C 1
0195 C C 10453
POWER DRIVE UNIT / HINGELINE GEAR BOX - SPOILER
2000 60 10 .0435 2
4 0 .5 1.8 1981
10 5.8 .1 0 0 .33
0190 C C 1
0195 C C 55
MOTOR - SPOILER
2000 60 5.0 .0296 2
4 0 .3 1.8 1981
5 5.3 .1 0 0 .33
0190 C C 1
0195 C C 55
INVERTER - SPOILER
2000 60 7 .0636 1
4 .3 .3 1.8 1981
6.3 5.52 .1 0 0 .33
51 6.941 .1 0 0 1
0190 C C 1
0195 C C 10453
```

TABLE B-2 RCA PRICE MODEL INPUT DATA-ALL-ELECTRIC AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 2 OF 5

```
00870   POWER DRIVE UNIT / HINGELINE GEAR BOX - LE FLAP
00880   3000 90 34.7 .1509 2
00890   6 0 .5 1.8 1981
00900   34.7 5.8 .1 0 0 .33
00910   0190 C C 1
00920   0195 C C 55
00930   MOTOR - L.E. FLAP
00940   3000 90 6.5 .0361 2
00950   6 0 .3 1.8 1981
00960   6.5 5.3 .1 0 0 .33
00970   0190 C C 1
00980   0195 C C 55
00990   INVERTER - L.E. FLAP
01000   3000 90 8.5 .0773 1
01010   6 .3 .3 1.8 1981
01020   7.6 5.52 .1 0 0 .33
01030   51 6.941 .1 0 0 1
01040   0190 C C 1
01050   0195 C C 10453
01060   BALL SCREW ACTUATOR - ENGINE INLET CENTERBODY
01070   1000 30 32 .1391 2
01080   2 0 .5 1.8 1981
01090   32 5.8 .1 0 0 .33
01100   0190 C C 1
01110   0195 C C 55
01120   MOTOR - ENGINE INLET CENTER BODY
01130   1000 30 5.0 .0296 2
01140   2 0 .3 1.8 1981
01150   5 5.3 .1 0 0 .33
01160   0190 C C 1
01170   0195 C C 55
01180   INVERTER -ENGINE INLET CENTER BODY
01190   1000 30 7.5 .0692 1
01200   2 .3 .3 1.8 1981
01210   6.7 5.52 .1 0 0 .33
01220   51 6.941 .1 0 0 1
01230   0190 C C 1
01240   0195 C C 10453
01250   POWER DRIVE UNIT/GEAR-BOX ENGINE INLET BYPASS DOOR
01260   2000 60 3 .013 2
01270   4 0 .5 1.8 1981
01280   3 5.8 .1 0 0 .33
01290   0190 C C 1
01300   0195 C C 55
01310   MOTOR- ENGINE INLET BYPASS DOOR
01320   2000 60 1 .0077 2
01330   4 0 .3 1.8 1981
01340   1 5.3 .1 0 0 .33
01350   0190 C C 1
01360   0195 C C 55
01370   INVERTER - ENGINE INLET BYPASS DOOR
01380   2000 60 1 .0091 1
01390   4 .3 .3 1.8 1981
01400   .9 5.52 .1 0 0 .33
01410   51 6.941 .1 0 0 1
01420   0190 C C 1
01430   0195 C C 10453
01440   BALL SCREW ACT-MAIN LANDING GEAR
01450   1000 30 20 .087 2
01460   2 0 .5 1.8 1981
01470   20 5.8 .1 0 0 .33
01480   0190 C C 1
01490   0195 C C 55
01500   MOTOR-MAIN &  NOSE LANDING GEAR
01510   1500 45 5 .0296 2
01520   3 0 .3 1.8 1981
01530   5 5.3 .1 0 0 .33
01540   0190 C C 1
01550   0195 C C 55
01560   INVERTER-MAIN & NOSE LANDING GEAR
01570   1500 45 5.7 .0519 1
01580   3 .3 .3 1.8 1981
01590   5.1 5.52 .1 0 0 .33
01600   51 6.941 .1 0 0 1
01610   0190 C C 1
01620   0195 C C 10453
```

TABLE B-2 RCA PRICE MODEL INPUT DATA-ALL-ELECTRIC AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 3 OF 5

```
01630   BALL SCREW ACT-NOSE LANDING GEAR
01640   500 15 20 .087 2
01650   1 0 .5 1.8 1981
01660   20 5.8 .1 0 0 .33
01670   0190 C C 1
01680   0195 C C  55
01820   ACTUATOR-NOSE GEAR STEERING
01830   500 15 20 .1 2
01840   1 0 .5 1.8 1981
01850   20 5.8 .1 0 0 .33
01860   0190 C C 1
01870   0195 C C 55
01880   MOTOR-NOSE GEAR STEERING
01890   500 15 4 .0248 2
01900   1 0 .3 1.8 1981
01910   4 5.3 .1 0 0 .33
01920   0190 C C 1
01930   0195 C C 55
01940   BULL RING ASSY-MAIN GEAR BRAKES
01950   1000 30 7 .0175 2
01960   2 0 .5 1.8 1981
01970   7 5.8 .1 0 0 .33
01980   0190 C C 1
01990   0195 C C 55
02000   MOTOR-MAIN GEAR BRAKES
02010   8000 240 .75 .0045 2
02020   16 0 .3 1.8 1981
02030   .75 5.3 .1 0 0 .33
02040   0190 C C 1
02050   0195 C C 55
02060   ROTARY ACTUATOR-AERIAL REFUELING DOOR
02070   500 15 8 .0348 2
02080   1 0 .3 1.8 1981
02090   8 5.8 .1 0 0 .33
02100   0190 C C 1
02110   0195 C C 55
02120   MOTOR-AERIAL REFUELING DOOR
02130   500 15 .25 .0021 2
02140   1 0 .3 1.8 1981
02150   .25 5.3 .1 0 0 .33
02160   0190 C C 1
02170   0195 C C 55
02180   LINEAR ACTUATOR-AERIAL REFUELING NOZZLE LATCH
02190   500 15 4 .0174 2
02200   1 0 .3 1.8 1981
02210   4 5.8 .1 0 0 .33
02220   0190 C C 1
02230   0195 C C 55
02240   MOTOR - AERIAL REFUELING NOZZLE LATCH
02250   500 15 .7 .0054 2
02260   1 0 .3 1.8 1981
02270   .7 5.3 .1 0 0 .33
02280   0190 C C 1
02290   0195 C C 55
02300   BALL SCREW ACTUATOR CANOPY
02310   500 15 7 .0304 2
02320   1 0 .3 1.8 1981
02330   7 5.8 .1 0 0 .33
02340   0190 C C 1
02350   0195 C C 55
02360   MOTOR - CANOPY
02370   500 15 1 .0077 2
02380   1 0 .3 1.8 1981
02390   1 5.3 .1 0 0 .33
02400   0190 C C 1
02410   0195 C C 55
02420   GEAR BOX - GUN DRIVE
02430   500 15 15.5 .0674 2
02440   1 0 .3 1.8 1981
02450   15.5 5.8 .1 0 0 .33
02460   0190 C C 1
02470   0195 C C 55
02480   MOTOR - GUN DRIVE
02490   500 15 11.2 .0541 2
02500   1 0 .3 1.8 1981
02510   11.2 5.3 .1 0 0 .33
02520   0190 C C 1
02530   0195 C C 55
```

TABLE 8-2 RCA PRICE MODEL INPUT DATA-ALL-ELECTRIC AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 4 OF 5

```
02540  INVERTER - GUN DRIVE
02550  500 15 9.8 .0871 1
02560  1 .3 .3 1.8 1981
02570  8.8 5.52 .1 0 0 .33
02580  51 6.941 .1 0 0 1
02590  0190 C C 1
02600  0195 C C 10453
02610  MOTOR-ECS BOOST COMPRESSOR
02620  500 15 21.4 .0839 2
02630  1 0 .3 1.8 1981
02640  21.4 5.3 .1 0 0 .33
02650  0190 C C 1
02660  0195 C C 55
02670  INVERTER-ECS BOOST COMPRESSOR
02680  500 15 19 .1727 1
02690  1 .3 .3 1.8 1981
02700  17.1 5.52 .1 0 0 .33
02710  51 6.941 .1 0 0 1
02720  0190 C C 1
02730  0195 C C 10453
02740  MOTOR-ECS PACK COMPRESSOR
02750  500 15 11 .05 2
02760  1 0 .3 1.8 1981
02770  11 5.3 .1 0 0 .33
02780  0190 C C 1
02790  0195 C C 55
02800  INVERTER - ECS PACK COMPRESSOR
02810  500 15 5.0 .0455 1
02820  1 .3 .3 1.8 1981
02830  4.5 5.52 .1 0 0 .33
02840  51 6.941 .1 0 0 1.0
02850  0190 C C 1.0
02860  0195 C C 10453
02870  MOTOR - ELECTRONICS COOLING
02880  500 15 18.4 .0760 2
02890  1 0 .3 1.8 1981
02900  18.4 5.3 .1 0 0 .33
02910  0190 C C 1
02920  0195 C C 55
02930  INVERTER - ELECTRONICS COOLING FAN
02940  500 15 16.0 .1455 1
02950  1 .3 .3 1.8 1981
02960  14.4 5.52 .1 0 0 .33
02970  51 6.941 .1 0 0 1.0
02980  0190 C C 1.0
02990  0195 C C 10453
03000  MOTOR / PUMP INVERTER COOLANT
03010  1500 45 2.5 .0167 2
03020  3 0 .3 1.8 1981
03030  2.5 5.8 .1 0 0 .33
03040  0190 C C 1
03050  0195 C C 55
03060  INVERTER - COOLANT PUMP
03070  1500 45 2.0 .0182 1
03080  3 .3 .3 1.8 1981
03090  1.8 5.52 .1 0 0 .33
03100  51 6.941 .1 0 0 1.0
03110  0190 C C 1.0
03120  0195 C C 10453
03130  RESERVOIR - INVERTER COOLANT
03140  1500 45 3.3 .033 2
03150  3 0 .3 1.8 1981
03160  3.3 5.52 .1 0 0 .33
03170  0190 C C 1
03180  0195 C C 53
03190  TUBING- INVERTER COOLANT
03200  14000 420 .789 .01 2
03210  28 0 .3 1.8 1981
03220  .789 5.7 .1 0 0 .33
03230  0190 C C 1
03240  0195 C C 53
```

TABLE B-2 RCA PRICE MODEL INPUT DATA-ALL-ELECTRIC AIRPLANE ACTUATION SYSTEMS, 500 AIRPLANES

SHEET 5 OF 5

```
HEAT EXCHANGER - INVERTER COOLANT
1500 45 2.0 .025 2
3 0 .3 1.8 1981
2.0 5.52 .1 0 0 .33
0190 C C 1
0195 C C 55
FILTER.WIRING, STRUCT. INST. - INVERTER COOLANT
1500 45 10.0 .125 2
3 0 .3 1.8 1981
10 5 .1 0 0 .33
0190 C C 1
0195 C C 33
INTEGRATION AND TEST COSTS
500 15 .7 .7 5
0 0 0 1.8 1981
0190 C C 0195 C
```

TABLE B-3 RCA PRICE MODEL INPUT DATA-BASELINE AIRPLANE SECONDARY POWER SYSTEM, 500 AIRPLANES

SHEET 1 OF 3

```
HIBA1.DAT  23-SEP-81  16:40:42

00100   **PRICE 84
00110   CYCLOCONVERTER          MODIFIED 9/4/81
00120   1000 30 60 .5455 1
00130   2 .5 .3 1.8 1981
00140   54 5.52 .1 0 0 .33
00150   51 6.941 .1 0 0 1
00160   0190 C C 1
00170   0195 C C 11153
00180   GENERATOR
00190   1000 30 30 .1734 2
00200   2 0 .3 1.8 1981
00210   30 5.3 .1 0 0 .33
00220   0190 C C 1
00230   0195 C C 55
00240   EMERGENCY GENERATOR
00250   500 15 36 .21 2
00260   1 0 .3 1.8 1981
00270   26 5.3 .1 0 0 .33
00280   0190 C C 1
00290   0195 C C 55
00300   HYD MOTOR-EMERGENCY GENERATOR
00310   500 15 14.8 .115 2
00320   1 0 .3 1.8 1981
00330   14.8 5.84 .1 0 0 .33
00340   0190 C C 1
00350   0195 C C 55
00360   CONTROL VALVE,ON-OFF - EMERGENCY GENERATOR
00370   500 15 1.09 .0045 1
00380   1 .5 .3 1.8 1981
00390   1.04 5.8 .1 0 0 .33
00391   40 7.94 .1 0 0 1
00392   0190 C C 1
00393   0195 C C 10055
00400   TRANSFORMER-RECTIFIER
00410   1500 43 12.5 .1136 1
00420   3 .5 .3 1.8 1981
00430   11.3 5.52 .1 0 0 .33
00440   51 6.941 .1 0 0 1
00450   0190 C C 1
00460   0195 C C 11153
00470   BATTERY,NI-CAD,24VDC,40AH
00480   500 15 75 .56 3
00490   1 0 .3 1.8 1981 1970
00500   0 0 0 0 75
00510   1.87 0 0
00530   BATTERY CHARGER
00540   500 15 6.8 .0618 1
00550   1 .7 .3 1.8 1981
00560   6.1 5.52 .1 0 0 .33
00570   51 6.941 .1 0 0 1
00580   0190 C C 1
00590   0195 C C 11153
00600   HYDRAULIC PUMP
00610   2000 60 27 .1125 2
00620   4 0 .3 1.8 1981
00630   27 5.84 .1 0 0 .33
00640   0190 C C 1
00650   0195 C C 55
00655   HYD RES #1
00660   500 15 11.5 .38 2
00670   1 0 .3 1.8 1981
00680   11.5 5.52 .1 0 0 .33
00690   0190 C C 1
00700   0195 C C 55
00710   HYDRAULIC RESERVOIR #2 & #3
00720   1000 30 5 .163 2
00730   2 0 .3 1.8 1981
00740   5 5.52 .1 0 0 .33
00750   0190 C C 1
00760   0195 C C 55
00830   RIGHT HAND AMAD GEARBOX
00840   500 15 99 .4 2
00850   1 0 .5 1.8 1981
00860   99 5.84 .1 0 0 .33
00870   0190 C C 1
00880   0195 C C 55
```

TABLE B-3 RCA PRICE MODEL INPUT DATA-BASELINE AIRPLANE
SECONDARY POWER SYSTEM, 500 AIRPLANES

SHEET 2 OF 3

```
00890   LEFT HAND AMAD GEARBOX
00900   500 15 93 .168 2
00910   1 0 .5 1.8 1981
00920   93 5.84 .1 0 0 .33
00930   0190 C C 1
00940   0195 C C 55
00950   ANGLE GEARBOX-AMAD
00960   500 15 39 .14 2
00970   1 0 .5 1.8 1981
00980   39 5.84 .1 0 0 .33
00990   0190 C C 1
01000   0195 C C 55
01010   POWER RELAY.AC.3PDT
01020   500 15 1.2 .018 2
01030   1 0 .3 1.8 1981
01040   1.2 5.7 .1 0 0 .33
01050   0190 C C 1
01060   0195 C C 55
01070   PWR CONTACTOR.AC 3PDT
01080   500 15 1.6 .0163 2
01090   1 0 .3 1.8 1981
01100   1.6 5.7 .1 0 0 .33
01110   0190 C C 1
01120   0195 C C 55
01130   PWR CONTACTOR.AC.3PST
01140   500 15 3.8 .0518 2
01150   1 0 .3 1.8 1981
01160   3.8 5.7 .1 0 0 .33
01170   0190 C C 1
01180   0195 C C 55
01190   PWR CONTACTOR.AC.3PST
01200   1500 45 5.3 .0636 2
01210   3 0 .3 1.8 1981
01220   5.3 5.7 .1 0 0 .33
01230   0190 C C 1
01240   0195 C C 55
01250   PWR CONTACTOR.AC.3PDT
01260   1000 30 6.2 .0708 2
01270   2 0 .3 1.8 1981
01280   6.2 5.7 .1 0 0 .33
01290   0190 C C 1
01300   0195 C C 55
01310   PWR CONTACTOR.DC.S?SY
01320   1500 45 .8 .009 2
01330   3 0 .3 1.8 1981
01340   .8 5.7 .1 0 0 .33
01350   0190 C C 1
01360   0195 C C 55
01370   PWR CONTACTOR.DC.SPDT
01380   1000 30 2.1 .0207 2
01390   2 0 .3 1.8 1981
01400   2.1 5.7 .1 0 0 .33
01410   0190 C C 1
01420   0195 C C 55
01430   HYDRAULIC TUBING
01440   14000 420 2.886 .0366 2
01450   28 0 .3 1.8 1981
01460   2.886 5.84 .1 0 0 .33
01470   0190 C C 1
01480   0195 C C 55
01485   ELECTRICAL WIRING
01490   14000 420 4.393 .0314 2
01500   28 0 .3 1.8 1981
01510   4.39 5 .1 0 0 .33
01520   0190 C C 1
01530   0195 C C 55
01540   INVERTER, STANDBY
01550   500 15 13 .1182 1
01560   1 .5 .3 1.8 1981
01570   12.7 5.52 .1 0 0 .33
01580   5A 6.941 .1 0 0 1
01590   0190 C C 1
01600   0195 C C 11153
01610   HYDRAULIC HAND PUMP
01620   500 15 3.4 .0309 2
01630   1 0 .5 1.8 1981
01640   3.4 5.56 .1 0 0 .33
01650   0190 C C 1
01660   0195 C C 55
01670   FLUID/FUEL HEAT EXCHANGER
01680   1500 45 3 .0375 2
```

TABLE B-3 RCA PRICE MODEL INPUT DATA-BASELINE AIRPLANE
SECONDARY POWER SYSTEM, 500 AIRPLANES
SHEET 3 OF 3

```
01690   3 0 .5 1.8 1981
01700   3 5.56 .1 0 0 .33
01710   0190 C C 1
01720   0195 C C 55
01730   TEMP CONTROL VALVE
01740   1500 45 1 .0042 1
01750   3 .5 .3 1.8 1981
01760   .95 5.8 .1 0 0 .33
01770   40 7.94 .1 0 0 1
01780   0190 C C 1
01790   0195 C C 10055
01800   OVER TEMP SWITCH
01810   1500 45 .1 .0009 2
01820   3 0 .3 1.8 1981
01830   .1 5.84 .1 0 0 .33
01840   0190 C C 1
01850   0195 C C 55
01860   RESERVOIR SERVICE PANEL
01870   500 15 10 .091 2
01880   1 0 .5 1.8 1981
01890   10 5.52 .1 0 0 .33
01900   0190 C C 1
01910   0195 C C 55
01920   PRESS/RETURN FILTER MODULE SYS 2.3
01930   1000 30 15 .1364 1
01940   2 .5 .3 1.8 1981
01950   14.5 3.8 .1 0 0 .33
01960   40 7.94 .1 0 0 1
01970   0190 C C 1
01980   0195 C C 10055
01990   PRESS/RETURN FILTER MODULE SYS 1
02000   500 15 23 .2091 1
02010   1 .5 .3 1.8 1981
02020   22.5 5.8 .1 0 0 .33
02030   40 7.94 .1 0 0 1
02040   0190 C C 1
02050   0195 C C 10055
02120   RESERVOIR BLEEDER VALVES
02130   3000 90 .1 .0004 1
02140   6 .5 .3 1.8 1981
02150   .09 5.8 .1 0 0 .33
02160   40 7.94 .1 0 0 1
02170   0190 C C 1
02180   0195 C C 10055
02190   RESERVOIR RELIEF VALVE (AIR & OIL)
02200   3000 90 .1 .0125 1
02210   6 .5 .3 1.8 1981
02220   .09 5.8 .1 0 0 .33
02230   40 7.94 .1 0 0 1
02240   0190 C C 1
02250   0195 C C 10055
02260   CASE DRAIN FILTER MODULE
02270   2000 60 8 .0727 2
02280   4 0 .5 1.8 1981
02290   8 5.56 .1 0 0 .33
02300   0190 C C 1
02310   0195 C C 55
02320   FIREWALL SHUTOFF VALVE
02330   2000 60 1.7 .0071 1
02340   4 .5 .3 1.8 1981
02350   1.42 5.8 .1 0 0 .33
02360   40 7.94 .1 0 0 1
02370   0190 C C 1
02380   0195 C C 10055
02390   SUCTION DISCONNECT
02400   2000 60 1.4 .0058 2
02410   4 0 .5 1.8 1981
02420   1.4 5.84 .1 0 0 .33
02430   0190 C C 1
02440   0195 C C 55
02450   HYD PRESS XMITTER
02460   1500 45 .2 .0018 2
02470   3 0 .3 1.8 1981
02480   .2 5.52 .1 0 0 .33
02490   0190 C C 1
02500   0195 C C 55
02570   GROUND SERVICE DISCONNECT
02580   3000 90 1.2 .005 2
02590   4 0 .5 1.8 1981
02600   1.2 5.84 .1 0 0 .33
02610   0190 C C 1
02620   0195 C C 55
02730   INT & TEST
02740   500 15 .7 .7 5
02750   0 0 0 1.8 1981
02760   0190 C C 0195 C
```

TABLE B-4 RCA PRICE MODEL INPUT DATA-ALL-ELECTRIC AIRPLANE SECONDARY POWER SYSTEM, 500 AIRPLANES

SHEET 1 OF 1

```
HIELI.DAT  24-SEP-81  15:45:02

00100   **PRICE 84
00110   STARTER GENERATOR                 MODIFIED 8/4/81
00120   1500 45 75 .4333 2
00130   3 0 .3 1.8 1981
00140   75 5.3 .1 0 0 .33
00150   0190 C C 1
00160   0195 C C 55
00170   PHASE DELAYED RECTIFIER BRIDGE
00180   1500 45 25 .2273 1
00190   3 .5 .3 1.8 1981
00200   22.5 5.52 .1 0 0 .33
00210   51 6.941 .1 0 0 1
00220   0190 C C 1
00230   0195 C C 11153
00240   DC-DC CONVERTER
00250   2000 60 17 .1545 1
00260   4 .5 .3 1.8 1981
00270   15.3 5.52 .1 0 0 .33
00280   51 4.941 .1 0 0 1
00290   0190 C C 1
00300   0195 C C 11153
00310   DC-AC INVERTER
00320   1000 30 34 .3091 1
00330   2 .5 .3 1.8 1981
00340   30.6 5.52 .1 0 0 .33
00350   51 6.941 .1 0 0 1
00360   0190 C C 1
00365   0195 C C 11153
00370   BATT-NICAD-24V-40AH
00380   1000 30 75 .54 3
00390   2 0 .3 1.8 1981 1970
00400   0 0 0 0 75
00410   1.87 0 0
00430   POWER CONTACTOR-SIX PHASE AC
00440   1000 30 18 .25 2
00450   2 0 .3 1.8 1981
00460   18 5.7 .1 0 0 .33
00470   0190 C C 1
00480   0195 C C 55
00490   PWR CONTACTOR- SIX PHASE AC(CROSS START)
00500   1000 30 12 .2 2
00510   2 0 .3 1.8 1981
00520   12 5.7 .1 0 0 .33
00530   0190 C C 1
00540   0195 C C 55
00550   PWR CONTACTOR-SINGLE PHASE AC
00560   2000 60 1 .0095 2
00570   4 0 .3 1.8 1981
00580   1 5.7 .1 0 0 .33
00590   0190 C C 1
00600   0195 C C 55
00610   PWR CONTACTOR-DC
00620   1500 45 9 .07 2
00630   3 0 .3 1.8 1981
00640   9 5.7 .1 0 0 .33
00650   0190 C C 1
00660   0195 C C 55
00670   PWR CONTACTOR-DC
00680   3000 90 6 .05 2
00690   6 0 .3 1.8 1981
00700   6 5.7 .1 0 0 .33
00710   0190 C C 1
00720   0195 C C 55
00730   WIRING AND CONNECTORS
00740   14000 420 8.25 .0589 2
00750   28 0 .3 1.8 1981
00760   8.25 5 0 0 0 .33
00770   0190 C C 1
00780   0195 C C 56
00790   INTEGRATION AND TEST
00800   500 15 .7 .7 5
00810   0 0 0 1.8 1981
00820   0190 C C 0195 C
```


*U.S.Government Printing Office: 1982 — 559-005/4069